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**LIGHTWEIGHT THERMAL PROTECTION SYSTEMS FOR  
SPACE VEHICLE PROPELLANT TANKS**

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**ABSTRACT**

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Efficient storage of propellants (especially cryogenics) in space for more than a few hours will generally require the use of multiple-foil insulation and/or shadow shields. The practical problems involved in applying both foil and shadow shields and the current state-of-the-art are discussed. Performance in both on Earth and a space environment are given for various foil insulated systems. Realistic weight estimates (supported by some experimental data) indicate that: (1) foil, despite its application penalties, outperforms all other non-foil insulations, and (2) shadow shield systems can be extremely lightweight and definitely should be considered for future thermal protection systems.

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## LIGHTWEIGHT THERMAL PROTECTION SYSTEMS FOR SPACE

### VEHICLE PROPELLANT TANKS

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### INTRODUCTION

Most space vehicles operating within both a planet's atmosphere and in the near vacuum of space will require some means of thermally protecting the propellant tanks. Depending upon the particular mission, it may be necessary to either minimize the amount of heat absorbed or the amount of heat lost by the propellant.

The methods available to control heat input or output are many, ranging from a simple insulated tank to the more complex machinery required for refrigeration. The choice of any particular method is usually dictated by weight considerations and the degree of complexity involved, since the complexity usually makes high reliability difficult to obtain. In general, thermal protection systems can be classified as passive or active; passive systems include surface coatings, conventional insulations, superinsulations, and shadow shields, whereas active systems include tank heaters, refrigeration systems, reliquification systems, and so forth. The passive methods generally provide the lightest weight and the least complex systems for the short duration (less than hours) and intermediate (greater than hours but less than years) missions. Active systems may be required for both cryogenic

*not of availability*

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and storable propellants on long duration (greater than years) missions or where extended planetary operations are necessary. The tradeoff between active and passive systems also depends on the surface area-to-volume ratio - the smaller volume tanks requiring active systems earlier than the larger tanks (Refs. 1 and 2).

This paper will concentrate on the thermal-protection systems for the intermediate duration missions because of both the immediate interest in this area (e.g., Apollo, planetary orbiters, and so forth) and the apparent gap in technology in the required thermal protection systems. The intermediate duration missions usually require the use of radiation barriers such as superinsulation (multiple-foil insulation) or shadow shields (Ref. 3 to 10). Radiation barriers, however, must be in a vacuum environment in order to be effective which consequently makes them incompatible with the necessary operations within the Earth's atmosphere. This is the primary source of problems encountered in applying lightweight radiation barriers to space vehicles.

It is the purpose of this paper to (1) briefly point out the parameters important to thermal protection, (2) examine the thermal protection methods, (3) discuss the application problems inherent in using foil insulation and review the current state-of-the-art, and (4) examine the problem and application areas for lightweight shadow shields.

#### HEATING ENVIRONMENT

In order to determine the type of thermal protection system required for any particular mission, the heat sources must first be examined. Heat sources can be classified as external and internal, external being those

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resulting from the vehicle's surrounding environment and internal being those resulting from adjacent vehicle components.

The significant external sources consist of the surrounding atmosphere, the sun, and the planets. Heating due to an atmosphere, both on Earth during prelaunch and aerodynamic heating due to passing through an atmosphere, can be severe and usually requires some means of thermal protection. The overall effect of atmospheric heating on the longer missions, however, can be made small by prelaunch and postlaunch operations (e.g., Refs. 11 and 12) such as replenishing boiloff from a cryogenic tank during prelaunch periods or possibly subcooling a propellant prior to launch and jettisoning the aerodynamic insulation after postlaunch heating.

Once outside of the atmosphere, the sun becomes the predominant source of heat. The maximum amount of heat that can be absorbed by a body at the Earth's distance from the sun is  $442 \text{ Btu/hr-ft}^2$ . This heat flux varies inversely with the square of the distance from the center of the sun, increasing to  $846 \text{ Btu/hr-ft}^2$  at Venus and decreasing to  $191 \text{ Btu/hr-ft}^2$  at Mars. The net amount of heat absorbed by a body at planetary distances from the sun is a function of the projected area, surface area, solar absorptivity, thermal emissivity, and temperature of the body (Ref. 13). To put this source of heat in proper perspective, consider the results of placing a stainless-steel (solar absorptivity = 0.54, thermal emissivity = 0.37) 10-foot-diameter spherical propellant tank at the Earth's distance from the sun. If the tank contained liquid hydrogen, heating from the sun alone would cause 100 percent of the propellant to boiloff per day (vented at 14.7 psia). The same tank with liquid oxygen would boil off 13 percent per day. Earth storable propellants such as UDMH and nitrogen

tetroxide would have boiloff rates of 0 and 2.1 percent per day, respectively. Hence, it is apparent that thermal protection from the sun must be provided for missions of any significant duration, especially for cryogenics.

Planetary heating results from reflected solar radiation (albedo heating) and direct thermal radiation from the planet (thermal heating). The radiation reaching a vehicle is a function of the altitude above the planet, the position with respect to the planet (e.g., night or day), and the attitude and geometry of the body in consideration (Refs. 14 to 18). For near-planetary operations, the planetary flux, although less than the solar flux, is usually sufficient to warrant propellant tank thermal protection. At altitudes greater than 10 planet radii, the planetary heating becomes negligible (Ref. 4). A very comprehensive treatment of planetary heating that includes both numerical and graphical results for several geometric configurations is given by Reference 14.

Heating from internal heat sources generally results from radiation and solid conduction from adjacent components such as payloads, warm rocket engines, and so forth. The heating or cooling can be severe depending on the particular mission and vehicle configuration. For example, conduction through supports and plumbing of a cryogenic tank can be acute for long duration missions.

Now that the heat sources have been defined briefly, the methods of thermal protection and the inherent application problems will be discussed. More extensive considerations of the heat sources are given in References 8, 14, and 15.

## THERMAL PROTECTION METHODS

### Coatings

Coatings are a form of surface treatment used to alter the absorptivity and emissivity characteristics of a surface. Since the predominant mechanism of heat transfer to a space vehicle is by thermal radiation, coatings provide a relatively inexpensive method, from the standpoint of weight considerations, of helping to control the heat input or output of a surface. For example, if the surface of a spherical hydrogen tank previously considered were coated with flat white acrylic paint ( $\alpha_s = 0.27$ , as given in Table I), the boiloff rate due to solar heating would be cut in half. The only penalty involved would be the weight of the coating used.

The surface coating also has a strong effect on the equilibrium temperature of a vehicle. Figure 1 gives the equilibrium temperature of perfectly insulated spherical and cylindrical surfaces rotating in the sun for various values of solar absorptivity to thermal emissivity ratios ( $\alpha_s/\epsilon_t$ ). Temperatures are given for surfaces at Venus, Earth, and Mars distances from the sun. The temperature of a particular surface is seen to vary roughly from  $100^\circ$  to  $400^\circ$  R traversing the distance between Venus and Mars. It is also quite apparent that the surface temperature can be reduced considerably with coatings. A clean 7075 aluminum surface, for example, can be reduced from  $790^\circ$  to  $400^\circ$  R by painting with flat white acrylic paint. For ratios around 1.0 (e.g., dull black enamel paint), Earth storable propellants could be stored indefinitely at the Earth's distance from the sun.

The equilibrium temperature as given in Figure 1 is very useful for preliminary analysis of thermal protection requirements in that it provides an upper boundary of the surface temperature for an insulated tank. The insulation requirements based on this temperature are quite conservative.

### Insulation

From the above discussion it is apparent that coatings alone cannot afford enough protection for the longer duration missions, especially when cryogenic propellants are used. The next obvious step is to insulate the propellant tank and to apply the coating on the outer surface of the insulation. For the 10-foot spherical hydrogen tank discussed previously, 1 inch of foam insulation would reduce the boiloff rate to 34.4 percent per day and 1 inch of foil insulation (assuming no deterioration in foil properties due to application) would reduce the rate to 0.105 percent per day. Hence, it is apparent that insulation can reduce the heat input considerably.

The remaining task is to select the insulation that will yield the lowest overall weight penalty for a given mission. For an insulated system where the heat input or output must be limited, the minimum weight penalty is obtained with the insulation with the lowest value of thermal conductivity times density ( $k\rho$ )\*. This applies, for example, when an insulation is used on a storable tank to prevent propellant freezing. For missions where a propellant will boiloff, and tradeoff must be made between

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\*See the appendix for the definition of all symbols.

the boiloff weight and the insulation weight. Without getting into too much detail about individual stage design, the weight penalty chargeable to a thermal protection system is given by

$$W_{tp} = Atp + \frac{KkA}{t h_v} \Delta T \tau \quad (1)$$

The first term is the insulation weight and the second is the boiloff weight penalty. The factor  $K$  accounts for the effect of losing boiloff before stage firing and the effect of additional structure required to contain the propellant that will boil off (e.g., Ref. 8). Solving the previous equation for the optimum insulation thickness and substituting back into equation (1) reveals that the minimum weight penalty is directly proportional to the  $\sqrt{kp}$ . Although the above analysis is grossly simplified, it serves to point out the insulation comparison parameter for systems where boiloff takes place. Table 2 lists values of  $k$  and  $p$  as well as the parameters  $kp$  and  $\sqrt{kp}$  for several insulations.

Of the nonevacuated insulations, foam appears most beneficial. If weight penalties for foam insulated systems are examined for missions of any significant duration, however, it immediately becomes apparent that foams are not satisfactory. Foam is currently being used on the Saturn SIVB, Saturn SII, and the Centaur, but all of these vehicles have short mission times (on the order of hours). Any extension of their mission times would result in large boiloff losses.

Other insulations in Table II with lower values of  $\sqrt{kp}$  include evacuated fiberglass, powder, and multiple foil insulations (Refs. 19 to 23). Of these insulations, multiple foil (more commonly called superinsulation) yields the lowest value of  $\sqrt{kp}$  and for this reason

has been advocated by most thermal protection studies in the past (e.g., Refs. 6, 8, and 9). Two examples of foil insulation are (1) Linde superinsulation (Ref. 21), which consists of alternate layers of aluminum foil and low conducting spacers (e.g., fiberglass), and (2) NRC-2 insulation (Ref. 23), which consists of mylar sheets with a vacuum deposit of aluminum on one side of each sheet (generally termed aluminized polyester). The thermal properties of the foil are usually expressed in terms of an apparent  $k$  that applies only for a given set of boundary conditions as shown in Figure 2 (from Ref. 6). These values are for vacuums less than  $10^{-5}$  Torr. The influence of vacuum, and, hence, the influence of gaseous conduction, on the thermal properties is shown in Figure 3 (from Ref. 24). These data are for helium gas and foil boundary temperatures of  $140^{\circ}$  and  $497^{\circ}$  R. It is apparent from this figure that pressures higher than  $10^{-5}$  Torr will seriously deteriorate the thermal properties of the foil insulation. This requirement of vacuum is the primary source of problems encountered in applying foil insulation to space vehicles that must necessarily start from the atmospheric environment of Earth. These problems will be discussed subsequently.

#### Shadow Shields

Another form of nearly passive thermal protection is provided by shadow shields or thermal radiation barriers. The shadow shields operate on the same basic principle as the foil insulation; the only difference being that the shadow shields are spaced further apart and, hence, radiate some of their heat to space. The equations for several forms of shadow shields are

given in References 8 and 25. In general, the heat transfer through a set of thermally isolated, infinitely conducting shadow shields is a function of the spacing of the shields, the emissivity and absorptivity, and the number of shields. This is illustrated in Figures 4 and 5.

Figure 4 shows the effect of placing  $N$  equally spaced shields between a room temperature source ( $530^{\circ}\text{R}$ ) and a simulated liquid hydrogen tank ( $37^{\circ}\text{R}$ ) for a range of spacing ratios using a constant value of emissivity and absorptivity of 0.1. The spacing ratio  $L_t/R$  is the distance between components divided by the radius of the components. It was assumed that there was heat transfer only between components (i.e., no heat transfer through the tank sides) and that the whole assembly was in a vacuum environment at  $0^{\circ}\text{R}$ . The effective temperature of space has been estimated to be roughly  $20^{\circ}\text{R}$ , (ref. 8) which is not high enough to significantly affect the results in Figures 4 to 7. The heat absorption rates given only apply for integer values of the abscissa. From Figure 4 it is apparent that a few shadow shields can reduce drastically the hydrogen tank heat absorption rates. For a spacing ratio of 1.0, the heat absorption rate with no shields is  $0.6\text{ Btu/hr-ft}^2$ . Placing two equally spaced shields between the bodies reduces the rate to  $0.0072\text{ Btu/hr-ft}^2$ ; with five shields, the rate becomes negative. The negative rates arise because the amount of heat reaching the tank through the shields is less than the amount of heat being radiated from the  $37^{\circ}\text{R}$  source to the absolute zero temperature of space. Figure 5 indicates that the absorptivity and emissivity ( $\alpha = \epsilon$ ) also have a strong effect on the heat absorption rate. For example,

reducing the emissivity from 1 to 0.1 decreases the heat transfer by two or three orders of magnitude, depending upon the number of shields.

Since the heat transfer is by radiation, it would be expected that the heat source temperature would have a strong effect on the heat transfer. This is shown on Figure 6. The heat absorption rate is shown versus source temperature for a range of shadow shields with a constant spacing ratio of 1.0 and a constant emissivity of 0.1. The significant fact to note about this figure is that the heat transfer from the very high temperature sources can be reduced considerably with a few shadow shields. For a  $3000^{\circ}\text{R}$  source, for example, the heat input is eliminated with 10 shields.

The shadow shields can also be used to shield a propellant from solar radiation. This requires some means of continuously pointing the shields toward the sun. For the solar shields, another variable is introduced and that is the surface properties of the outermost shield- namely, the solar absorptivity and thermal emissivity. The effects of the surface properties are shown in Figure 7 where the heat transferred into a hydrogen tank is given as a function of number of shadow shields for a spacing ratio of 0.5; that is, the outermost shield is located 0.5 radii from the end of the propellant tank. The shields are in a 3-degree conical array to allow some latitude in the attitude control system that orients the shields toward the sun. The figure indicates that it is desirable to have a low solar absorptivity and a high thermal emissivity as expected. It is interesting to note that even with the small spacing ratio used, the heat transfer into a hydrogen tank can be eliminated with nine shields. Larger spacing ratios could eliminate the heat transfer with fewer shields (e.g., Fig. 4).

From the preceeding discussion, it is apparent that shadow shields are by far the most efficient means of reducing heat transfer to a space vehicle propellant tank providing the radiation from the heat source is unidirectional. The decision whether or not to use shields depends on the particular mission and vehicle configuration restraints. Shadow shields alone, for example, could not effectively protect a propellant tank from both the sun and a planet during near-planetary operations.

#### Interconnecting Structural Members and Plumbing

Another important area of thermal protection lies in the proper selection and design of interconnecting structures and plumbing. This is especially important for cryogenic tankage. For the purposes of this paper, the penalty chargeable to the interconnecting members is

$$W_{ps} = W_{structure} + CW_{bo} \quad (2)$$

where  $C$  is a constant to account for boiloff prior to stage firing.

Grossly simplified, the weight penalty can be written as follows:

$$W_{ps} = \frac{Pl\rho}{S} + \frac{CP_r}{\lambda h_v} \int_{T_1}^{T_2} \frac{k dT}{S} \quad (3)$$

where the load over the stress  $P/S$  replaces the area of the structural member. For a given vehicle it is quite likely that  $P$ ,  $C$ ,  $r$ ,  $h_v$ , and possibly  $\lambda$  will be fixed. This reduces the weight penalty to a function of the material density-to-stress ratios  $\rho/S$  and the product of thermal conductivity and temperature difference over the stress  $\int \frac{k dT}{S}$ . The integral is used because  $k$  varies considerably between room temperature

and cryogenic temperatures. For the long duration missions, the  $\int k dT/S$  parameter is the overpowering factor in the total weight penalty. The density-to-stress ratio  $\rho/S$  will probably be a second order effect because the interconnecting structural members and plumbing are usually a small percentage of the vehicle structural weight.

Again, although the preceeding equations are approximate, they are useful in that they point out the important parameters involved. These parameters are shown in Table III for several materials. Examination of Table III will immediately point out that of the metals considered, aluminum, the most commonly used structural member, will cause the largest boiloff penalties. Using titanium will reduce the penalty (represented by  $\int k dT/S$ ) by a factor of 20. The nonmetals, despite their relatively poor structural properties, should also be considered for the long duration missions as they have the lowest values of  $\int k dT/S$ . The total weight penalties due to heat leaks through struts and plumbing depend greatly on the vehicle and the mission duration. Values for a typical vehicle are given subsequently.

The thermal protection methods discussed thus far, that is, coatings, insulations, shadow shields, and low conducting interconnecting members, are the most obvious passive means available for thermal protection on the intermediate range missions. Additional methods to help reduce the thermal protection system weights, which involve varying degrees of complexity, are available. Some of these are the selective placement of components to give a monotonic temperature gradient, the utilization of boiloff gases for cooling heat sources or warm insulation surfaces, the utilization of the

heat sink available in propellants (this may require a mixing device since propellants will probably stratify in a zero gravity condition), and the selection of heating environment (e.g., moon operations on dark side of moon for cryogenics).

#### APPLICATION PROBLEMS

The purpose of this section will be to concentrate on the application problems of foil insulation and shadow shields because of the potential weight savings available with these systems over the conventionally insulated systems for the intermediate duration missions. This is not to imply, however, that there are no application problems inherent in the other thermal protection methods discussed.

Coatings, in general, will provide a straightforward method of controlling the heat input (over an order-of-magnitude range) for the short duration missions. Extended missions times, however, will require determining the effects of meteoroid erosion, prolonged solar radiation, and surface evaporation on the surface characteristics of the materials. Reference 26 gives the effects of the space environment on various materials.

Conventional insulations for cryogenic tankage will provide some problems for both the short and intermediate range missions (e.g., Ref. 27) such as sealing the insulation to prevent cryogenic pumping or purging the insulation with a noncondensable gas and applying relatively large thicknesses of insulation. For most cryogenic missions over a few hours duration, the weight penalties will be prohibitive unless special steps are taken to minimize the heat reaching the propellant tank (use of shadow shields, e.g.).

Design of tank supports, necessary plumbing, and so forth for cryogenic tanks will be influenced heavily by thermal considerations for intermediate and long duration missions. Both low conducting metallic and nonmetallic materials and unconventional design techniques (Ref. 28) will have to be employed to minimize the penalties due to structural heat leaks.

#### Foil Insulation

Application methods and problems. - As mentioned previously, the vacuum requirement for foil insulation makes it incompatible with the necessary operations within the atmosphere unless special steps are taken to provide a vacuum. Ground storage dewars, with their heavy double-walled vacuum jackets, have used multiple foil insulation for years because weight is not of primary importance. Obviously, these systems would be quite impractical for most space vehicles. Consequently, a lightweight method is required for applying foils to a space-weight propellant tank to accommodate the transition from an Earth environment to a space environment and still retain the desirable insulating properties of the foil. This is probably the single most important problem facing the foil application field today.

Other application problems exist including (1) designing the system to avoid boost aerodynamic loading and heating of the foil, (2) determining methods of minimizing heat leaks due to penetrations and discontinuities in the insulation (Ref. 28), (3) determining manufacturing techniques of applying the insulation to space vehicle tankage (Ref. 29), and (4) determining the necessary handling techniques. These problems are important

but will be of no interest unless a reasonable application method is determined to insure efficient foil performance during or shortly after the Earth-space environmental transition.

There are, in general, two methods available to circumvent the heavy double-walled dewar structure normally used to provide a vacuum environment for foils. The most obvious method is to build a specially designed lightweight vacuum jacket for the propellant tank. The other method consists of sacrificing ground performance by allowing gases within the foils and then using the vacuum of space for foil evacuation.

Recent efforts to build a lightweight vacuum jacket have resulted in the vacuum-bag concept (Ref. 21, 30, 31, and 32). The vacuum-bag technique (Fig. 8(a)) consists of laying the foil insulation directly on the tank surface and then enclosing the whole assembly within a flexible vacuum-tight bag. A vacuum is drawn between the vacuum bag and tank wall which causes an atmospheric compression load on the foil insulation. The compression load in turn causes an increase in the heat-transfer rate, which will correspondingly decrease as the propellant tank moves up through the atmosphere into the vacuum of space. Small scale vacuum-bag tests (Ref. 21) first proved the feasibility of this concept and generally gave encouraging results. The tests indicated that the insulation would recover 95 percent of its original thickness and maintain its original value of thermal conductivity once the atmospheric compression load was removed (entire assembly placed in a vacuum chamber). Failure of the insulation to recover its original thickness was attributed to permanent sets in the vacuum bag. Heat-transfer rates during atmospheric compression were roughly 100 times higher than the normal uncompressed rates. The compressed rates

agree well with the results of the well-controlled laboratory tests reported in Reference 33 that also indicated a factor of 100. References 30 and 31 reported the results of applying a vacuum bag (1/2 mil mylar - 1 mil Al - 1/2 mil mylar) to the cylindrical section of a 1.5-foot diameter, 9-foot-long tank insulated with foil. The ends of the cylinder were insulated by other means. The compressed insulation heat-transfer rate during a simulated ground hold was still roughly 100 times the normally uncompressed rate, but after release of the atmospheric force, the insulation only recovered 72 percent of its original thickness. Heat-transfer rates through the decompressed insulation were not available, but they are expected to be considerably higher than those reported for the small-scale tests.

Recent work (Ref. 32) on a 4-foot-diameter insulated hydrogen tank more closely simulated an actual space vehicle application. The tank was completely insulated with 1 inch of foil insulation and then enclosed in a vacuum bag. Again the compressed heat-transfer rates were roughly 100 times the normally uncompressed rates. The heat-transfer rate of the decompressed insulation (space simulation) was 0.8 Btu/hr-ft<sup>2</sup>, which is roughly seven times higher than that predicted using ideal foil properties for uninterrupted insulation. These rates correspond to an effective thermal conductivity of  $1.66 \times 10^{-3}$  Btu in./(hr)(ft<sup>2</sup>) °R for the 1 inch of foil. The density of the installed insulation was 11.7 lb/ft<sup>3</sup> (includes vacuum bag, and so forth) which yields  $k_p$  and  $\sqrt{k_p}$  values of 0.0194 and 0.139, respectively. These results are encouraging in that they represent a reasonable simulation of an insulated propellant tank.

The deterioration of the foil properties in the vacuum-bag tests were, as mentioned earlier, largely a result of permanent sets in the vacuum-bag material. This is also supported by the fact that compression tests of insulation samples alone (with no vacuum bag) indicate no deterioration of the thermal properties after compression and decompression (Ref. 24). The degree of vacuum-bag recovery will depend greatly on the resiliency of the underlying insulation. The tests reported in References 21, 30, 31, and 32 used alternate layers of aluminum foil and fiberglass mat which will probably be more resilient than the crinkled aluminized polyester insulation. Methods of improving the vacuum-bag recovery, which will in turn improve foil performance, include (1) using a thinner more flexible bag material, (2) removing the bag once in the environment of space, (3) using a special means of "popping" the bag out to its original shape once in space, and (4) providing a means of suspending the bag from the foil insulation surface to avoid compression.

Assuming the bag recovery problem can be solved, there still remains the problem of avoiding leaks in the vacuum jacket during atmospheric operation. Good seals around struts and plumbing, for example, must be maintained especially on cryogenic tankage to avoid condensation of in-leaking gases within the foil insulation. It should be noted that leaks in the vacuum bag may be extremely hard to detect due to condensation of the in-leaking gases and the initially high heat-transfer rates of the compressed foil. Hence, it would be possible to unknowingly launch a system with a seriously deteriorated insulation unless an adequate vacuum monitoring system were available. Also, any leaks and subsequent gas condensation within the

insulation would require extremely long hold times for repair. A method of circumventing the gas condensation problem is shown in Figure 8(b). Another insulation such as foam is attached to the tank wall to provide a relatively warm base surface for the foil (Ref. 29). The surface temperature of the foam is high enough to avoid condensation of incoming gases. The foam requirements for such a system are shown in Figure 9. A film heat-transfer coefficient of  $2 \text{ Btu/hr-ft}^2$  OR and an environmental temperature of  $540^\circ \text{ R}$  were used to obtain these curves for a foil-foam combination on a liquid hydrogen tank. The hydrogen tank was vented at 14.7 psia. The heat transfer through the compressed foil under one atmosphere of pressure was assumed to be 100 times that of the uncompressed insulation (Ref. 30 and 33). The results indicate that the foam required to maintain a  $160^\circ \text{ R}$  base surface is roughly equivalent to the uncompressed foil thickness. Also plotted in Figure 9 is the boiloff rate with and without base insulation. It is apparent that the base insulation contributes little to decreasing the boiloff rates. The added weight of the foam varies from  $0.20$  to  $0.88 \text{ lb/ft}^2$  for  $1/4$  inch and 3 inches of foil, respectively (includes  $0.10 \text{ lb/ft}^2$  glue line and  $0.03 \text{ lb/ft}^2$  seal). This weight, however, is traded for the increased reliability of the integrated thermal protection system. The surface of the foam would, of course, have to be adequately sealed to prevent contamination of the foils by outgassing.

The second major method of applying foils to propellant tanks involves the use of purge gases during atmospheric operation with the subsequent evacuation of these gases in the vacuum of space (Ref. 34 and 35). The purged foil technique (Fig. 8(c)) is much the same as the vacuum-bag concept

with the exception that the space between the bag and tank wall is purged with a noncondensable gas. For a liquid hydrogen tank, a helium purge in the insulation would be required, whereas nitrogen could be used for a liquid oxygen tank. Storable, or room temperature, propellants require no purges. Provisions must be made to rapidly evacuate this system once in space because, as indicated in Figure 3, the apparent thermal conductivity of the foil is a strong function of vacuum, and long evacuation times consequently lead to large heat inputs or outputs. The time required to evacuate a foil-insulated system using the vacuum of space is unknown at this time. Experimental results of the vacuum-bag tests reported in References 30 and 32, however, indicate that once a system is sufficiently out-gassed, the pressure within the foil insulation can be reduced from atmospheric to 5 to  $50 \times 10^{-3}$  Torr in a few hours. These tests were with limited ground pumping equipment. Using the vacuum of space will certainly result in shorter evacuation times. It should also be noted that the gas flow area was considerably restricted in these tests due to both the presence of the fiberglass mats between the foils and the compression of the insulation during evacuation of the vacuum bag. A purged foil system will not be subjected to compression forces and will probably utilize the aluminized polyester insulation, hence providing a relatively unrestricted flow area for the exiting purge gas. Methods have been suggested to hasten evacuation of foil insulated systems (Ref. 36), including perforating the foils, but this usually deteriorates the properties as well. Since no one has attempted to evacuate a purged system for a simulated propellant tank, it is premature to discuss augmentation methods at this time.

Another problem that exists in the purged system is the possibility of "blowing" the foil off during rapid decompression on ascent through the atmosphere (Ref. 24). This can be avoided, however, with careful design and use of a containment bag. Again, as with purge gas evacuation, this is a relatively unexplored area and requires some experimental research.

The performance of the purged foil during ground hold will be considerably deteriorated due to the presence of the purge gas. The apparent thermal conductivity of the foil insulation will be at least equivalent to the thermal conductivity of the purge gas itself (Ref. 30). Using this assumption, the boiloff rates of several purge systems are shown in Figure 10 as a function of insulation thickness. For a hydrogen tank with a helium purge, the boiloff rate for 1 inch of insulation is  $1.2 \text{ lb/hr-ft}^2$ . This is about a factor of twenty higher than the rates obtained with a comparable vacuum-bag system during ground hold (Fig. 9). These higher boiloff rates are not necessarily intolerable, though, as evidenced by the fact that Centaur also uses a helium purge during ground hold.

Also shown in Figure 10 is the effect of using different purge gases for an oxygen tank. The lower thermal conductivity of the  $\text{N}_2$  gas results in much lower boiloff rates than those for the helium gas, which immediately points out the latitude available with the selection of purge gases. In order to use a low thermal conductivity purge gas and, hence, realize lower ground losses, it must be insured that the surfaces over which it passes are above the gas condensation temperature. For the hydrogen tank, this is done by adding a base insulation to furnish the necessary temperature

increase (Fig. 8(d)). Figure 11 is included to demonstrate the boiloff rates of such a system for various purge gases ( $\text{CO}_2$ , A,  $\text{N}_2$ , and Ne) as functions of foil thickness. In each case, the thickness of the base insulation (foam, for this example) is sufficient to raise its surface temperature above the condensation temperature of the particular purge gas. From the figure it is seen that  $\text{CO}_2$  gives the lowest boiloff rate (an order-of-magnitude less than a helium purged system), but requires considerably more base insulation than the other gases. If boiloff during ground hold is considered secondary to total insulation weight, then neon would be the most desirable purge gas, assuming all other things equal, because it gives the lowest foam insulation thickness and boiloff rates a factor of three less than the helium purge. However, higher weight penalties due to the base insulation will probably be accepted in order to use the relatively inexpensive and plentiful purge gas nitrogen. The boiloff rates for the nitrogen curve lie between the rates for the helium purge system and the vacuum-bag system. The weight penalty for the required foam varies from 0.15 to 0.30 lb/ft<sup>2</sup> for 1/4 and 3 inches of foil, respectively. It is interesting to note that as a purge gas (e.g., nitrogen) is removed, the thermal conductivity of the foil will decrease considerably which will in turn cause the temperature of the base insulation to decrease. The decreased surface temperature of the base insulation may be sufficient to cryopump the foil and, hence, hasten obtainment of a good vacuum.

The basic application methods discussed thus far, that is, vacuum bag, purge bag, and vacuum or purge bag with an additional base insulation, must be protected from aerodynamic forces and heating. For most space

missions of any duration, the propellant tanks will be "buried" either by meteoroid or aerodynamic shields and, hence, will avoid this problem. Tanks directly exposed to aerodynamic heating will impose larger weight penalties for foil application due to the necessary protection required for the foils. A few methods of providing this protection are shown in Figure 12. They include an outer reinforced metallic or nonmetallic (e.g., foam or cork) vacuum jacket, a metallic or nonmetallic shield (for aerodynamic protection only) with purged insulation (Ref. 34 and 35), and a purged foil held in compression by an outer layer of jettisonable insulation. Whether these systems can be used or not depends on the success of the vacuum- and purge-bag concepts. The latter concept (Fig. 12(c)) will be quite desirable if purge gases can be evacuated rapidly in space. The outer insulation, reinforced foam, for example, could be banded to the tank and jettisoned in space. The purged foils would then tend to "fluff-out" as the purge gas escapes and, hence, provide an uncompressed high-vacuum insulation.

A common problem area in foil application for both vacuum and purged systems is the effect of penetrations and discontinuities in the insulation. In applying foil insulation to a propellant tank, penetrations in the insulation are required for structural support members, fill and vent tubes, and insulation seams. These provide areas where additional heat can enter both by radiation and by conduction. Considerable analytical effort has been placed in the area of penetrations and discontinuities, including the work reported in References 28, 29, 37, and 38. The analytical approach has thus far indicated that significant heat leaks will result unless extreme

care is exercised in applying the foil. Several suggested methods of minimizing these heat leaks are given in References 29 and 38. Analytical models have been devised for various types of penetrations and discontinuities (Refs. 28 and 37), and experimental verification of these models is now underway (Ref. 24). Although the results of this work are not yet available, related experimental work completed thus far (Ref. 32 and 39) indicates that the heat-leak contributions due to penetrations and discontinuities in the insulation will not cause an order-of-magnitude increase in the heat flux through an insulated propellant tank. The heat flux for the insulated tank reported in Reference 32, which included realistic insulation seams and one major penetration, was roughly seven times higher than that predicted for an ideal insulation (continuous blanket with no penetrations, and so forth). This seven-fold increase in heat flux included performance deteriorations due to both the incomplete vacuum-bag recovery and the  $0.5 \times 10^{-3}$  Torr vacuum as well as deteriorations due to penetrations and discontinuities. The work reported in Reference 39 determined the performance of a 3-foot-diameter spherical tank insulated with aluminized polyester insulation. The tank included penetrations for supports and fill and vent lines as well as the necessary insulation seams. Under simulated space conditions (high vacuum), the resulting heat leak through the insulation was roughly 25 percent higher than that predicted using ideal foil properties. Although there was considerable scatter in the data presented, these results (and those in Ref. 32) indicate that the heat leak contributions due to penetrations and discontinuities will be considerably less than an order-of-magnitude effect.

The remaining problem area with foil application involves the manufacturing and handling techniques for large foil insulated propellant tanks. These techniques will depend largely on whether a vacuum- or purge-bag system is used. For example, the foil for a purge-bag system will have to be more secure than for a vacuum-bag system because of the decompression forces. On the other hand, the handling requirements for a vacuum-bag system will be more stringent in order to avoid puncturing the vacuum bag. Punctures in a purge bag should not be a serious problem. Minimization of the effects due to penetrations and discontinuities will be common to any system as will the methods of applying the foils to cylindrical, spherical, toroidal, and other shaped tanks.

Summarizing, both the vacuum-bag and purge-bag technique appear to be reasonable methods of applying foil to space-weight tankage. For cryogenic tanks, either system can be augmented in reliability or performance with the addition of a base insulation. Application of foil to storable propellant tanks will present fewer problems because the possibility of condensation in the foil disappears. Research is needed on (1) methods of improving vacuum-bag recovery, (2) evacuation rates of purge gases, (3) mechanical integrity of foil during rapid decompression, (4) sealing of base insulations, (5) base insulation requirements for both systems, (6) effects of penetrations and discontinuities, and (7) manufacturing and handling techniques of large foil insulated tanks.

Weight penalties for foil insulated systems. - At the present time, it is difficult to say which of the foil application concepts discussed will result in the lowest overall weight penalty for a given mission. In order

to summarize the weight penalties associated with foil application the effects on the  $k\rho$  and  $\sqrt{k\rho}$  comparison parameters are needed. An attempt at this is shown in Figure 13 for a liquid hydrogen tank (sufficient for relative comparisons). The  $\sqrt{k\rho}$ , which gives a direct comparison of the weight penalties for a vented hydrogen tank, is shown for various systems as a function of uncompressed foil thickness. The outer insulation temperature was assumed to be  $530^{\circ}\text{R}$  for all cases. Curve 1 on Figure 13 is for foil with no application penalties; that is, a blanket of insulation with ideal values of thermal conductivity and density. Foam (curve 8) and evacuated perlite (curve 9) are also with no application problems. Curves 2 through 7 demonstrate the penalties involved for various applications of foil. These curves include weights of necessary seals ( $0.03\text{ lb/ft}^2$ ), vacuum or purge bags ( $0.03\text{ lb/ft}^2$ ), base insulation ( $3\text{ lb/ft}^3$ , foam in this case), glue line for base insulation ( $0.10\text{ lb/ft}^2$ ), and foil weight ( $4.7\text{ lb/ft}^3$ ). Penalties due to penetrations and discontinuities in the insulation are not included because so much depends on the particular installation method. As indicated previously, however, the resulting heat leak will be much less than an order-of-magnitude effect which in turn means that the  $\sqrt{k\rho}$  values presented will increase considerably less than a factor of three over the idealized foil (curve 1).

The penalty for a vacuum bag with perfect recovery can be obtained by comparing curves 1 and 2. If a base insulation is added beneath the foil (curve 3) to enhance the system reliability, the  $\sqrt{k\rho}$  parameter increases by a factor of 1.5 above the case with foil and a vacuum bag. The results of an actual vacuum-bag system, as reported in Reference 32, are depicted

by curve 4. The point at 1 inch is an experimental point where the rest of the curve is extrapolated. A major portion of the increase of the  $\sqrt{kp}$  is probably due to the incomplete vacuum-bag recovery. The remaining performance deterioration is a result of the high density of the installed insulation (11.7 lb/ft<sup>3</sup>), the vacuum ( $0.5 \times 10^{-3}$  Torr), the heat leaks through insulation seams, and so forth. The results are very encouraging though, because the performance is still about four times better than foam on a  $\sqrt{kp}$  basis.

The weight penalties that might be incurred in purged foil systems are depicted by curves 2 and 5 to 7. The effect of a deteriorated vacuum is shown by curves 2, 5, and 6. Even with a poor vacuum of  $10^{-3}$  Torr (curve 6), the purged system shows weight penalties about a factor of eight less than the foam. The relative effect of adding foam and using a nitrogen purge (to reduce ground losses) instead of a helium purge is obtained by comparing curves 2 and 7. Both of these curves assume the residual purge-gas pressure is less than  $10^{-5}$  Torr. Decreasing the ground boiloff rates by a factor of 5 (from Fig. 10 and 11) in this case causes a 30 percent increase in weight penalty.

Although penalties are incurred for all application methods, they are considerably less than the nearest nonvacuum insulation competitor foam. It appears that evacuated powders will also cause higher weight penalties, especially since they are faced with the same vacuum requirement as the foil. From the curves it is apparent that increasing the reliability of the vacuum-bag concept by adding foam increases the weight penalties, but the penalties still are not unreasonably high and, hence, this concept should

definitely be considered. The penalties due to poor evacuation of purged systems also do not appear severe, indicating that this is a very worthwhile concept to pursue.

### Shadow Shields

Application methods and problems. - Shadow shields, as a means of protection from the sun, have received little consideration in many mission studies because of orientation requirements or near-planetary operations. They have usually been ruled out along with the conventional insulations (e.g., foam) in favor of the foil insulation. After considering the many foil application problems, however, it might be wise at this point to reassess the shadow shield as a means of thermal protection for these missions.

The performance of shadow shields depicted in Figures 4 to 7 was based upon the assumptions of thermally isolated, infinitely conducting shields and a vacuum low enough to negate gaseous conduction. Of these assumptions only the latter can be definitely attained, and then only in the vacuum environment of space.

The assumption of thermally isolated, infinitely conducting shields, of course, cannot be obtained. With careful design of the shield supporting structure and proper spacing of the shields, however, the desirable properties of the shadow shields can be realized. Again, as with the foil insulation, the performance of the shadow shields could deteriorate considerably and still provide a very effective method of limiting heat input. Design of supporting structure for the shields must follow the same principles used to limit heat transfer through plumbing and tank supports (Refs. 28 and 30).

Typical shadow shield supports are shown in Figure 14. In these designs every attempt was made to provide poor thermal contact between the shield and supporting member. In practice, this should not be too difficult since the shadow shields will generally be extremely light, a few pounds or less.

The assumption of infinitely conducting shields could be approximated closely by using relatively thick aluminum shields. This adds weight to the system, however, and must be weighed against the penalties resulting from using thin low conducting shields. Work in Reference 40 on dual shadow shields indicates temperature variations will occur across zero conducting shields for low values of spacing ratio ( $L_t/R = 0.1$ ). For larger spacing ratios on the order of one, the temperature variations are minimized. It is not known at this time, however, how the spacing ratio will affect the heat transfer through two or more shields.

The requirement of a vacuum environment for shadow shields cannot be met while in the atmosphere. The nature of the shadow shields is such, however, that evacuation in space is no problem. This is because the shields are far enough apart to permit almost instantaneous evacuation once in the vacuum of space. Elaborate protection methods for the shadow shields, therefore, are not required. Additional insulation to limit excessive ground losses may be required, but this will in no way affect the shield performance. The insulation for ground hold could be foam or perhaps purged foil depending upon the mission.

In summary, most of the work to date on shadow shields has been analytical in nature. Both experimental research and further analytical research is needed on (1) the effect of spacing ratio, absorptivity (solar and thermal),

and emissivity on shadow shield heat transfer, (2) the effect of conduction between the shields and supporting structure, (3) the effect of using realistic shield conductivities, and (4) the inherent vehicle integration problems.

Weight penalties for shadow shield systems. - The weight penalties chargeable to a shadow shield system depend on the particular mission, but, in general, can include one or more of the following: shield weight, support weight, additional structure required to accommodate the shields (due to space needed for reasonable spacing ratios), additional insulation required for ground protection, and attitude control system weight. To optimize a system for cryogenic protection, these weights (less the attitude control system) must be traded against boiloff weight. The resulting tradeoff is not as straightforward as the insulation tradeoffs that were made on a  $\sqrt{k\rho}$  basis. For this reason direct comparison of foil insulated systems against shadow shield systems cannot be made without a detailed design study.

In order to obtain a relative comparison of thermal protection systems, both foil and shadow shields are considered for a single burn hydrogen-oxygen upper stage that places a payload in a Venus orbit. The stage is placed on a transfer trajectory to Venus with a Cl-B Centaur. Mission time (storage time) is 120 days from Earth orbit to Venus orbit, and the stage weighs roughly 4500 pounds with a propellant loading of 4100 pounds. The resulting thermal protection systems are shown in Figures 15 and 16. Both vehicles are purged with cold helium gas prior to launch and jettison aerodynamic insulation after leaving the atmosphere.

For the foil insulated system, it was assumed that (1) the outer foil temperature was constant at 420° R (Fig. 1 for  $\alpha_s/\epsilon_t = 0.3$ ), (2) the propellant boils off at 14.7 psia, (3) the K factor given in equation (1) is  $1.08/e^{(\Delta V/I_g)}$  as given in Reference 8, and (4) ideal foil properties are used (curve 1 of Fig. 13). With these assumptions, the following was obtained:

Propellant tank	Optimum foil thickness	Foil weight	Boiloff Weight	Payload penalty
Hydrogen	1.1 in.	65 lb	136 lb	132 lb
Oxygen	1.6 in.	49 lb	99 lb	98 lb

These weight penalties, about 230 pounds for the entire system, are for perfect foil properties. The penalties for a more realistic system are obtained by ratioing the  $\sqrt{kp}$  parameters in Figure 13. For example, using the experimental results reported in Reference 32 (data point on Figure 13), the total payload penalties will be about four times higher which could negate the usefulness of this stage for this particular mission. As a matter of interest, the propellant boiloff due to heat transfer through the plumbing and structure is roughly 34 and 71 pounds for the hydrogen and oxygen tanks, respectively.

The shadow shield design (Fig. 16) assumed (1) constant orientation toward the sun, (2) outer shield temperature of 530° R, (3) inner shields have absorptivities and emissivities of 0.1, (4) propellant boils off at 14.7 psia, and (5) ideal shadow shield performance. Using these assumptions, the weight penalties of the shielded vehicle shown in Figure 16 are as

follows: (1) shadow shield weight - 23 lb, (2) additional support structure - 18 lb, (3) shield attachments - 20 lb, and (4) boiloff penalty - 10 lb.

Thus, the total weight penalty less the attitude control system is 71 pounds. The ground boiloff rates could be reduced by adding 0.5 inch of foam to the tanks, which adds another 47 pounds, bringing the total to roughly 118 pounds. These weight estimates are crude, but are of the right order of magnitude. If the vehicle spends any time in Earth orbit additional penalties will result. The boiloffs due to conduction through the supports are 10 and 39 pounds for the hydrogen and oxygen, respectively.

Although the results of both systems (foil and shadow shields) are admittedly optimistic, they are useful in that they serve to point out the large gains available with the use of shadow shields. For the mission used, the weight penalty for the shadow shield system was roughly one half of that for the foil insulated system. These gains will also magnify for longer duration missions. The only weight penalty not included for the shield system was that of the attitude control system. For many missions the attitude control system may be required for other reasons (e.g., maintaining a stabilized vehicle to enable communications with Earth via high gain antennas) and will not be chargeable to the shadow shield system. Even if the attitude control system is considered as a weight penalty, shadow shields should definitely be considered for most long duration missions. This is especially true in light of the additional penalties possible with foil application.

Shadow shields will also provide very lightweight thermal protection systems whenever high-temperature components are on board or when near

passes to the sun are made. The latter case is interesting in that two shadow shields with the capability of varying the spacing ratio from 0 to 1.0 (extendable shields) can completely eliminate the heat transfer into a 530° R payload (manned perhaps) to distances within 0.4 AU of the sun.

As mentioned earlier, the shadow shields are practically useless for cryogenic protection near the planets. However, even here they may serve a useful function in that they eliminate the largest heat source, the sun. Future thermal protection systems, then, will probably use combinations of shadow shields and foils and, hence, utilize the desirable properties of both. For example, on missions to nearby planets, the propellant tanks could be effectively protected by shadow shields during interplanetary transfer and by foil during orbital operations. This would allow long time periods for outgassing before the foil is needed.

### CONCLUSIONS

Comparisons of available thermal protection systems for propellant tankage indicates that multiple foil insulation and/or shadow shields will provide the least weight penalties for the intermediate duration space missions (hours to years). Application problems exist for both foil and shadow shields, but do not appear insurmountable.

Two of the more promising methods of applying foil insulation to "buried" space tankage are the vacuum-bag and the purge-bag concepts. Considerable experimental effort has been placed on the vacuum-bag concept with encouraging results. The reliability of the vacuum-bag appears

questionable, however, due to the possibility of condensing in-leaking gases. Addition of a base insulation beneath the foil will considerably enhance the system reliability. Little effort has been placed on the purge-bag concept despite its inherent advantages in simplicity and reliability. This concept should definitely be pursued further as it may very well provide the lightest method of applying foil. Analytical and some experimental results indicate that either concept will out-perform all other nonfoil insulations despite deteriorations and compromises in the foil properties. Experimental research is needed on vacuum-bag recovery, sealing of base insulations, evacuation rates of purge gases, mechanical integrity of foil during rapid decompression, base insulation problems and requirements, effects of penetrations and discontinuities, manufacturing techniques of applying foil to large space vehicle tankage, and the inherent handling problems.

Research is also needed on methods of applying foils to surfaces that will be exposed to aerodynamic loads and heating. Although the weight penalties will be higher than that of the "buried" tankage, the foil insulated systems will still weigh less than the conventionally insulated systems (e.g., foam) for missions more than a few hours in duration.

Application of shadow shields to space vehicle tankage should be less of a problem than foil application, primarily because no special provisions have to be made for the transition from an Earth environment to a space environment. Care must be taken, however, in the shadow shield support methods used in order to insure proper thermal communication between the supporting structure and shields. For cryogenic tankage, additional insulation

(foam, foil, and so forth) may be required to reduce excessive ground losses. Research is needed on the effects of conduction between shields and supporting members and the effects of shield thermal conductivity, spacing ratio, absorptivity, and emissivity on the shield temperatures and resulting heat transfer.

The choice between a foil insulated system and a shadow shield system depends on the particular vehicle and mission. For missions requiring extended near-planetary operations, foils will yield the lightest weight system. Conversely, missions requiring long coast periods in interplanetary space will require the use of shadow shields. Shadow shields will also provide the lightest thermal protection systems whenever intense heat sources are encountered - for example, near passes to the sun. For many missions, however, the desirable properties of both foil and shadow shields can be used to complement each other. Long duration missions to nearby planets could be protected effectively by shadow shields during interplanetary transfer and by foils during orbital operations around a planet. Shadow shield protection during the trip to a planet, or the moon, could allow the foil insulation many hours or days to outgas and, hence, become very effective.

APPENDIX - SYMBOLS

A	surface area, $\text{ft}^2$
C	constant to account for boiloff prior to stage firing
g	gravitational constant, $32.2 \text{ ft/sq sec}$
$h_f$	film coefficient, $\text{Btu/hr-sq ft } ^\circ\text{R}$
$h_v$	heat of vaporization, $\text{Btu/lb}$
I	specific impulse, sec
K	factor to account for the effect of losing boiloff before stage firing and the effect of additional structure required to contain the propellant that will boil off
k	apparent mean thermal conductivity of insulation, $\text{Btu in./hr-sq ft } ^\circ\text{R}$
$L_T/R$	total length of shadow shield system/radius of cold body
l	length of structural supports, lb
N	number of shadow shields or foils
P	load on structural supports, lb
S	stress of structural supports, $\text{lb/sq in.}$
t	thickness of insulation, in.
$T_{\text{AMB}}$	ambient temperature, $^\circ\text{R}$
$T_h$	temperature of hot surface, $^\circ\text{R}$
$W_{\text{bo}}$	boiloff weight, lb
$W_{\text{ps}}$	total weight penalty chargeable to tank structural members, lb
$W_{\text{tp}}$	total weight penalty, lb

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$\alpha_t$	thermal absorptivity of surface
$\alpha_s$	solar absorptivity of surface
$\epsilon$	total hemispheric emissivity of surface
$\epsilon_t$	total hemispheric emissivity of surface exposed to sun
$\rho$	insulation density, lb/cu ft
$\sigma$	Stefan-Boltzman constant, $1.713 \times 10^{-9}$ Btu/hr-sq ft $^{\circ}\text{R}^4$
$\tau$	time, hr
$\Delta T$	temperature difference, $^{\circ}\text{R}$
$\Delta V$	velocity increment, ft/sec

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TABLE I. - ABSORPTIVITIES AND EMISSIVITIES OF MATERIALS AT ROOM TEMPERATURE

Finish	References	Solar Absorptivity, $\alpha_s$	Emissivity, $\epsilon_t$	Solar absorptivity to thermal emissivity ratio, $\alpha_s/\epsilon_t$
Aluminum 0.01 oxide	41	0.21	0.82	0.26
Flat white acrylic paint	41	.27	.895	.30
Dull black enamel	41	.91	.85	1.07
347 Stainless steel	41	.54	.27	2.0
Clean 7075 aluminum	41	.55	.11	5.00
410 Stainless heated to 1300° F Smooth	3	.76	.124	6.13
Sand-blasted		.796	.428	1.86
Rokide coated 410 stainless heated to 1300° F	3	.252	.80	0.315
Lampblack	3	.97	.96	1.01

TABLE II. - COMPARISON OF VARIOUS INSULATIONS

Insulation	Thermal Conductivity between 520° R and 37° R, $k$ , Btu in./hr-ft <sup>2</sup> or	Density, $\rho$ , lb/ft <sup>3</sup>	$k\rho$ , Btu in. hr-ft <sup>2</sup> or ft <sup>3</sup>	$\sqrt{k\rho}$ , Btu in. hr-ft <sup>2</sup> or ft <sup>3</sup> ) <sup>1/2</sup>
Corkboard	30,000x10 <sup>-5</sup>	20	6.0	2.45
Fiberglass	21,000	4	.84	0.92
Foam	14,800	2	.296	.54
Evacuated perlite	864	8	.069	.262
Evacuated fiberglass	1,000	4	.040	.20
Evacuated foil	24	4.7	.001125	.0338

TABLE III. - MATERIAL PROPERTIES

Material	Density, $\rho$ , lb/ft <sup>3</sup>	Stress, $S$ , lb/in. <sup>2</sup>	$\int_{160}^{520} k \, dT$ Btu/hr-ft	$\int_{160}^{520} k \, dT/S$ , in. <sup>2</sup> /ft <sup>3</sup>	$\rho/S$ , in. <sup>2</sup> /ft <sup>3</sup>
Aluminum	168	50,000	18,800	0.376	0.00336
Titanium	281	93,000	1,334	.01435	.00302
Stainless	484	90,000	2,540	.02824	.00538
Dacron	---	18,000	31.7*	.00176	-----
Nylon	71	18,000	64.8*	.0036	.00710
Teflon	137	1,800	50.4*	.0280	.0761
Kel-F	131	7,500	12.6*	.00168	.01746
Reinforced fiberglass	121	28,000	60.0*	.00214	.00432

\*Values obtained using room temperature values of thermal conductivity (should be conservative).

## FIGURE LEGENDS

Fig. 1. - Equilibrium temperature of a perfectly insulated sphere and cylinder rotating in the sun.

Fig. 2. - Apparent thermal conductivity of Linde SI insulation between variable ambient temperature and liquid hydrogen and liquid oxygen temperatures. Data from Reference 8.

Fig. 3. - Effect of helium gas pressure on apparent thermal conductivity of foil insulation. Data from Reference 29.

Fig. 4. - Effect of number of shadow shields and spacing ratio on heat transfer between two bodies,  $\alpha = \epsilon = 0.1$ .

Fig. 5. - Effect of emissivity and number of shadow shields on the heat transfer between two bodies,  $L_T/R = 0.1$ .

Fig. 6. - Effect of source temperature and number of shields  $N$  on heat transfer between two bodies,  $L_T/R = 1.0$ ,  $\alpha = \epsilon = 0.1$ .

Fig. 7. - Effect of outer surface solar absorptivity and thermal emissivity on heat transfer through conical array of shields,  $L_T/R = 0.5$ ,  $\alpha = \epsilon = 0.1$ .

(a) Foil enclosed in vacuum bag. (b) Insulation combination with vacuum bag.

Fig. 8. - Various methods of applying foil insulation to cryogenic tanks.

(c) Foil enclosed in purge bag. (d) Insulation combination with purge bag.

Fig. 8. - Concluded. Various methods of applying foil insulation to cryogenic tanks.

Fig. 9. - Effect of uncompressed foil thickness on hydrogen boiloff rate and required thickness of foam.

Fig. 10. - Effect of purge gas and foil insulation thickness on boiloff rates.

Fig. 11. - Effect of foil thickness on  $H_2$  boiloff rate and required foam thickness for various purge gases.

Fig. 12. - Foil insulated tanks capable of withstanding aerodynamic loads and heating.

Fig. 13. - Effect of various insulation systems on the insulation comparison parameter.

Fig. 14. - Low conducting shadow shield supports.

(a) Boost from Earth.      (b) Earth-Venus transit.

Fig. 15. - Venus orbiter with foil insulation.

(a) Boost from Earth.      (b) Earth-Venus transit.

Fig. 16. - Venus orbiter with shadow shields.

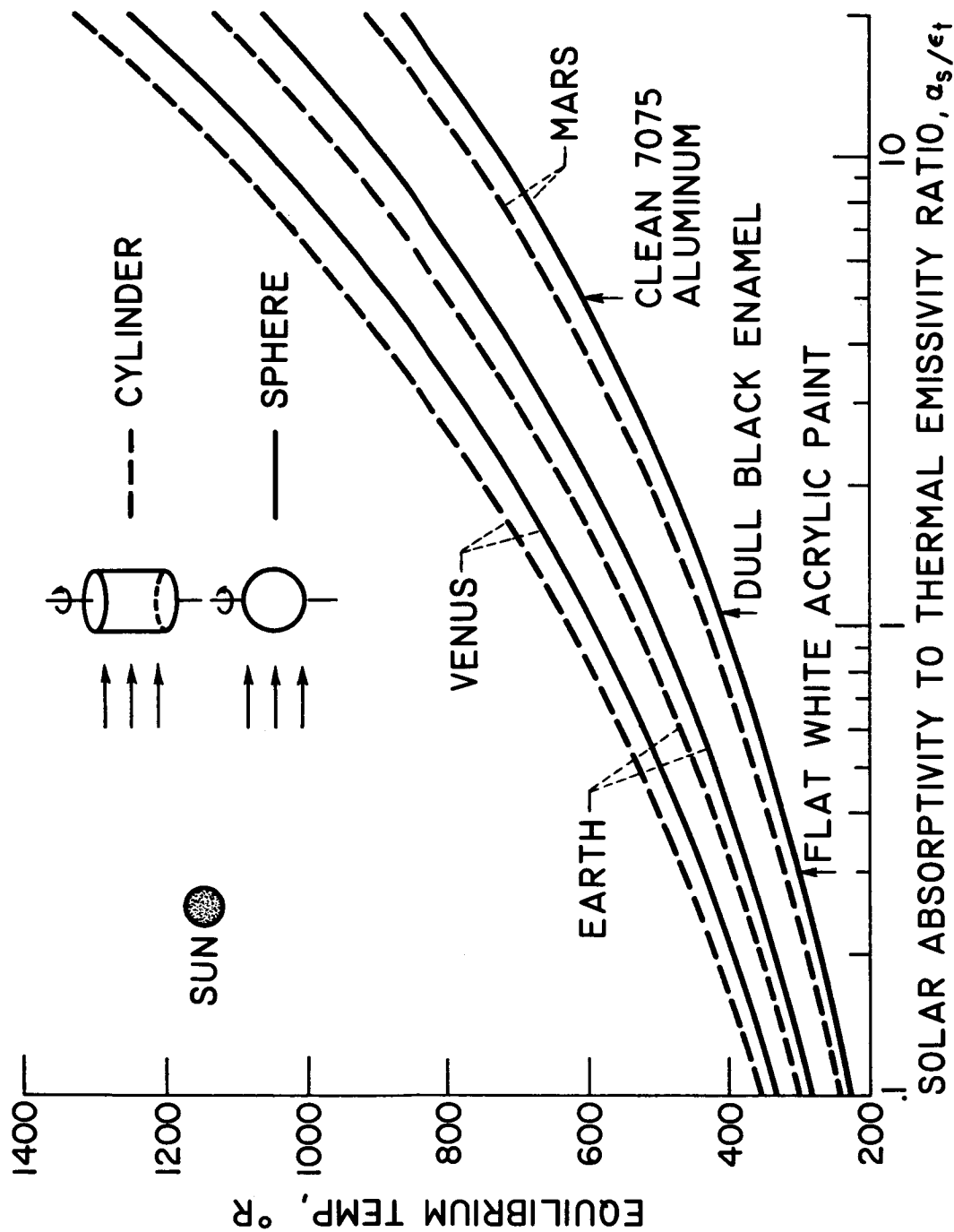


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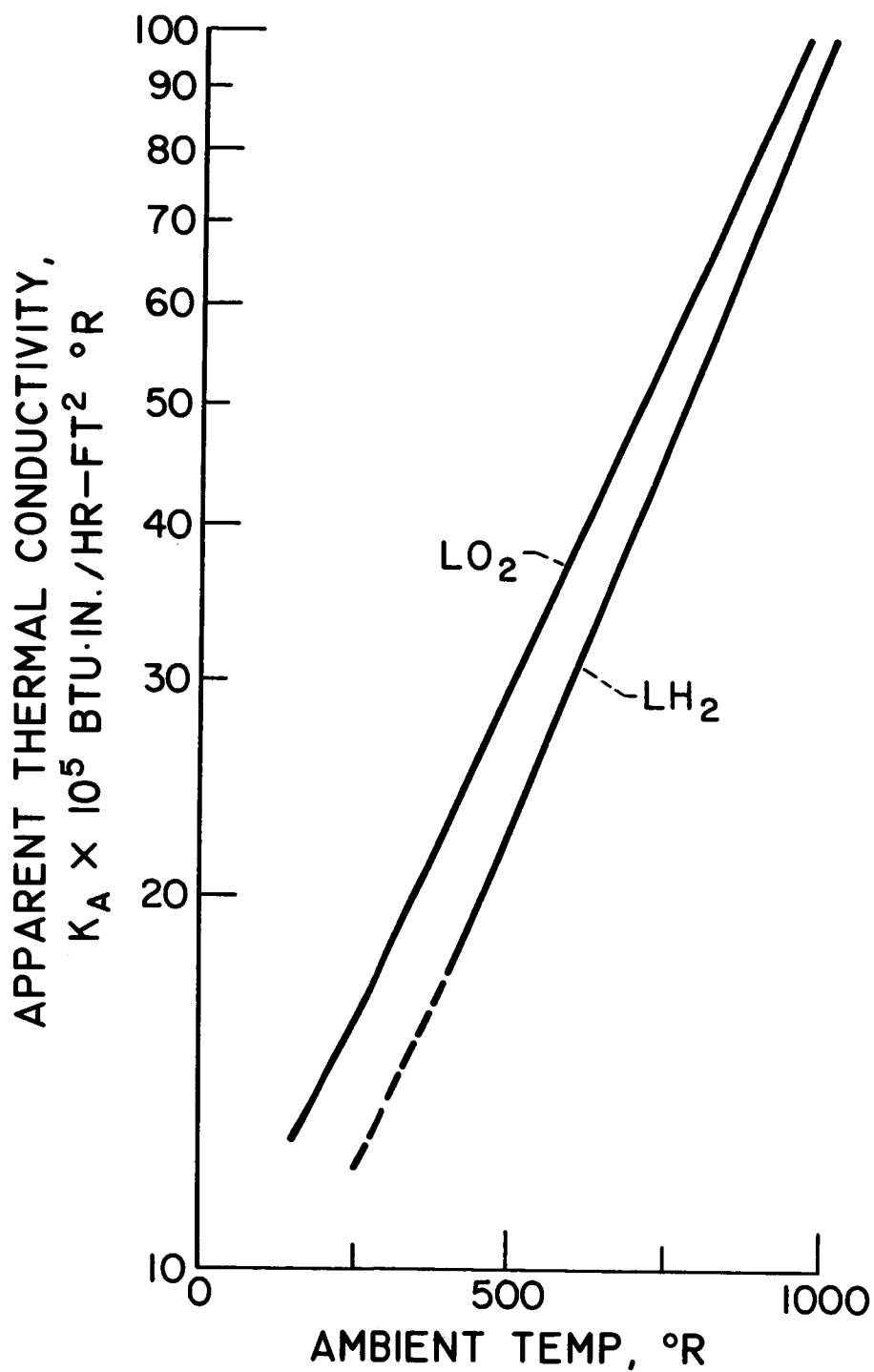


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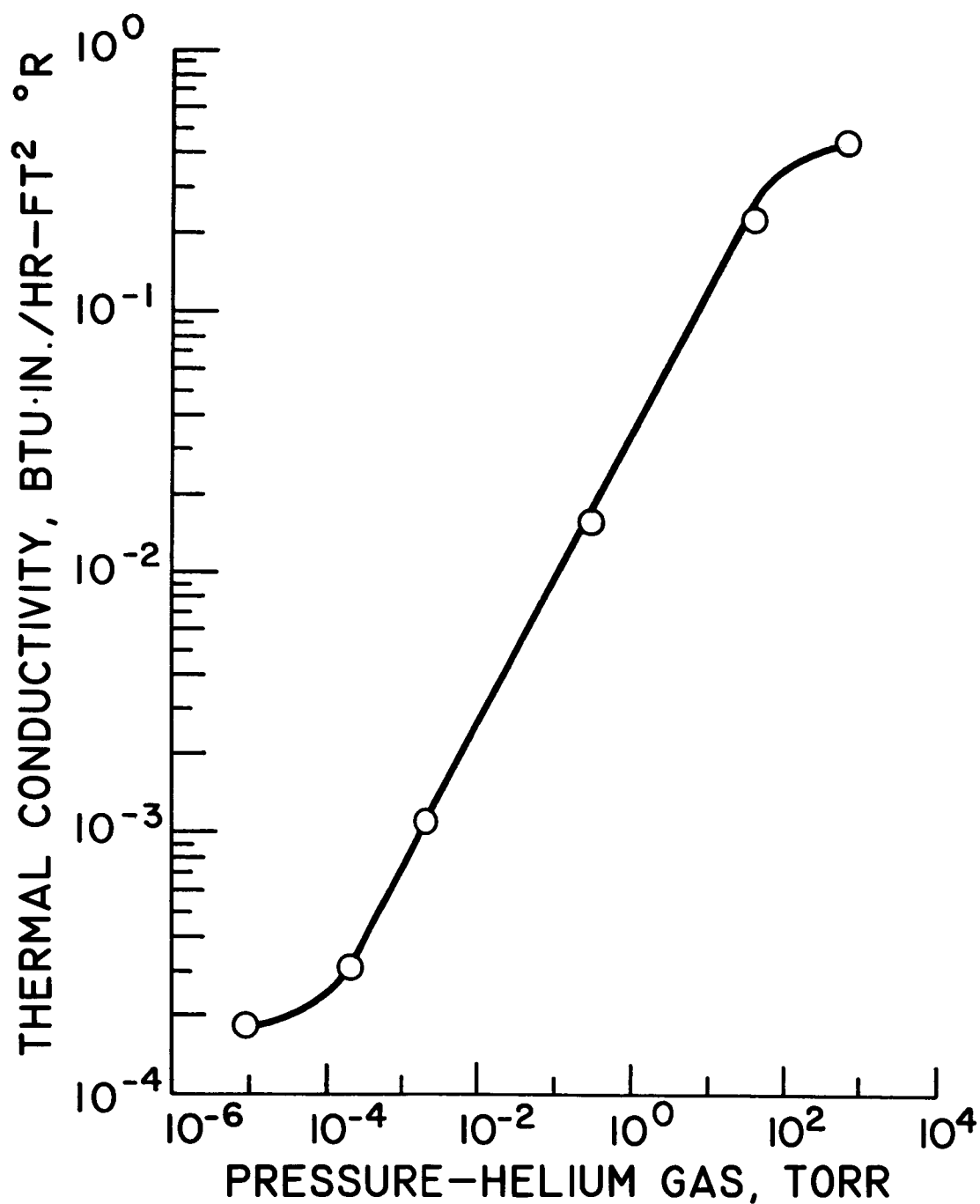


Fig. 3. - Effect of helium gas pressure on apparent thermal conductivity of foil insulation. Data from Reference 29.

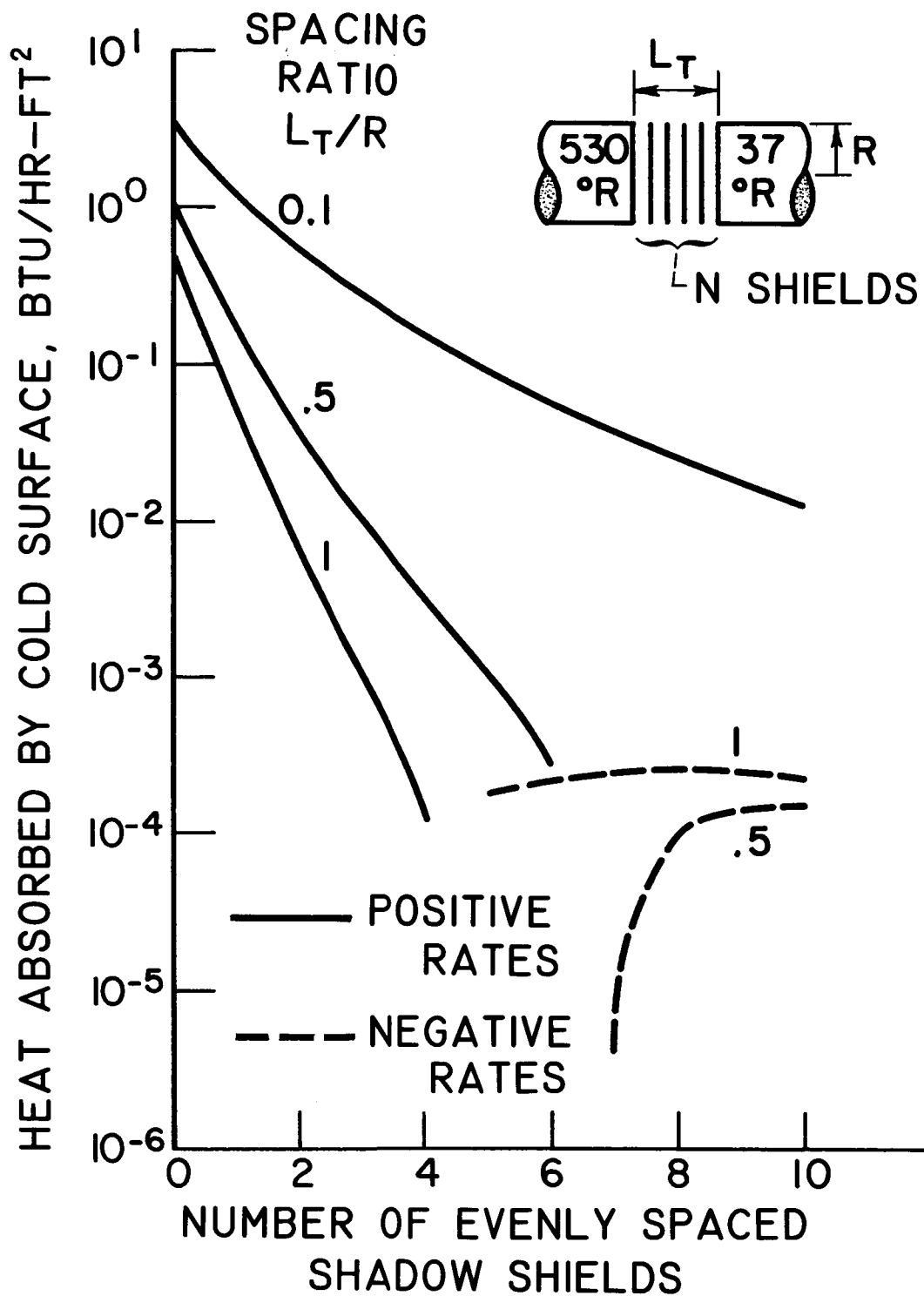


Fig. 4. - Effect of number of shadow shields and spacing ratio on heat transfer between two bodies,  $\alpha = \epsilon = 0.1$ .

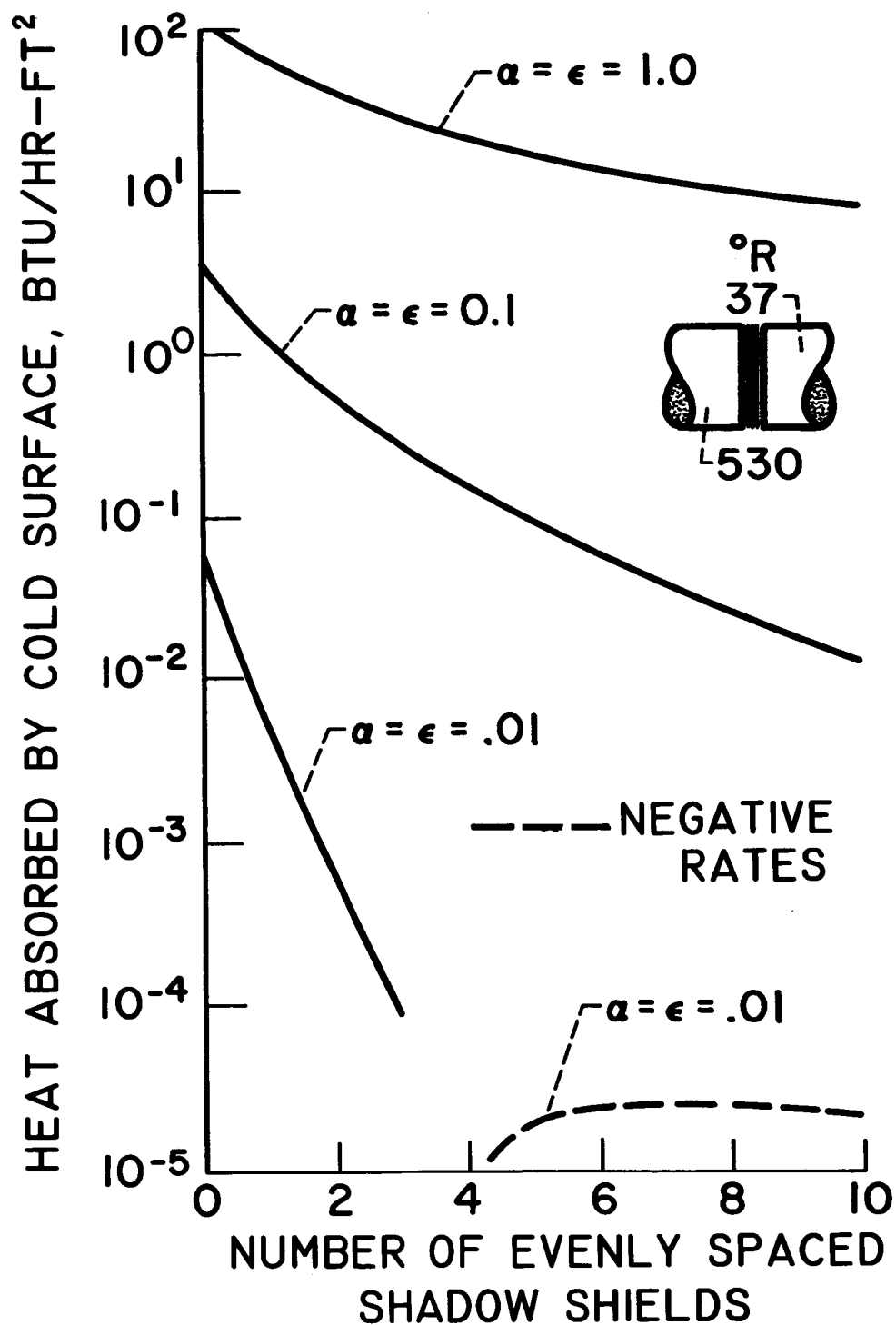


Fig. 5. - Effect of emissivity and number of shadow shields on the heat transfer between two bodies,  $L_T/R = 0.1$ .

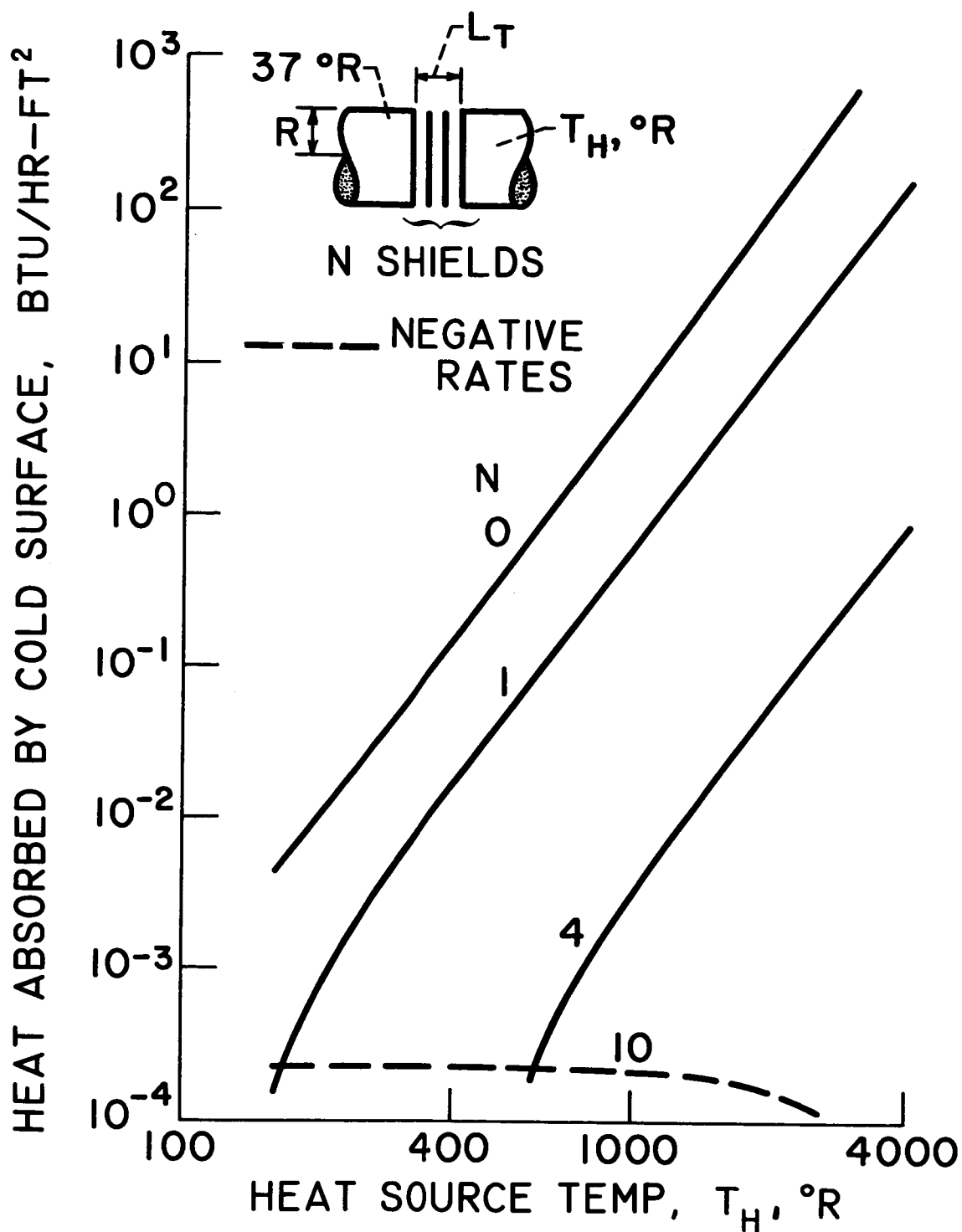


Fig. 6. - Effect of source temperature and number of shields  $N$  on heat transfer between two bodies,  $L_T/R = 1.0$ ,  $\alpha = \epsilon = 0.1$ .

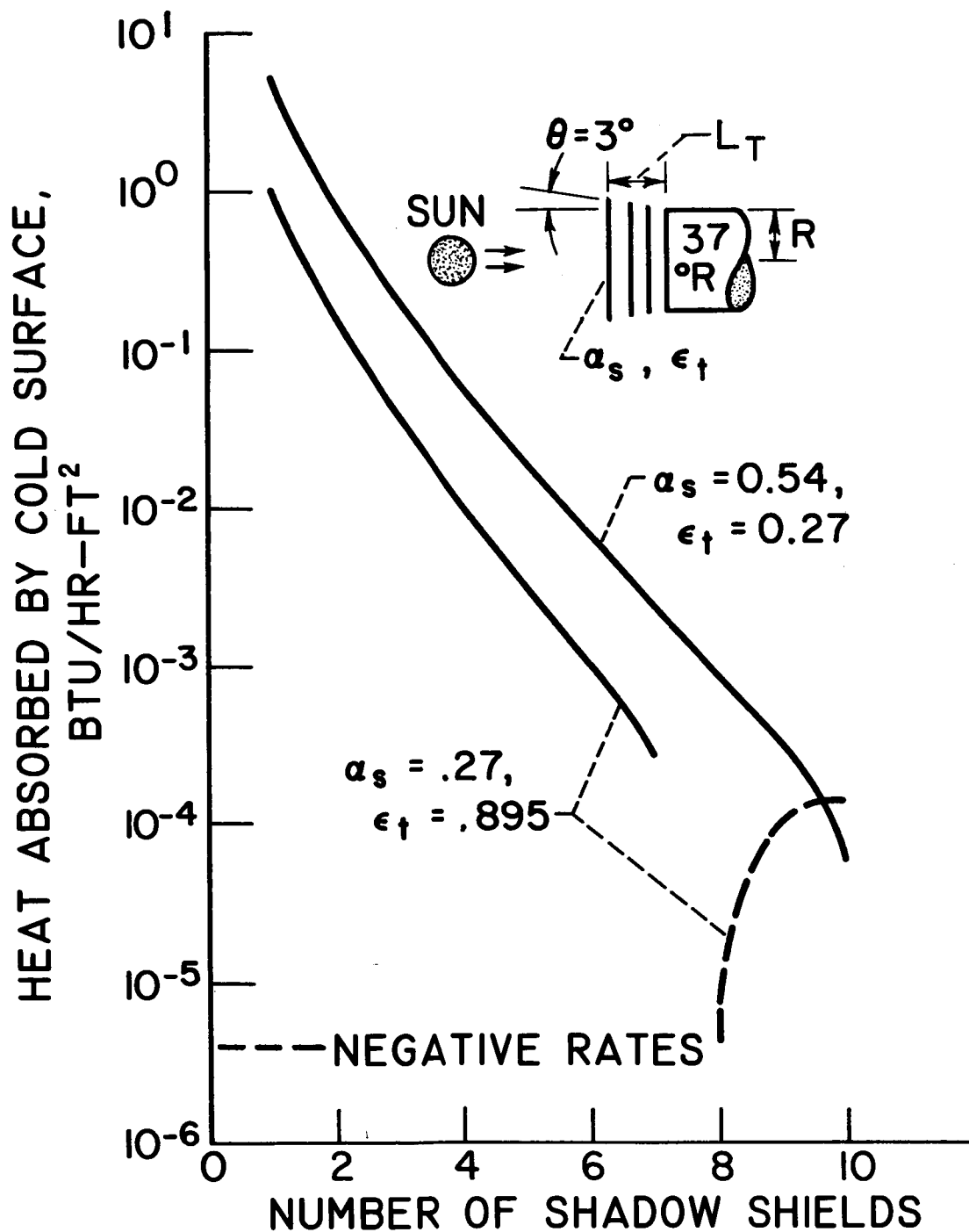
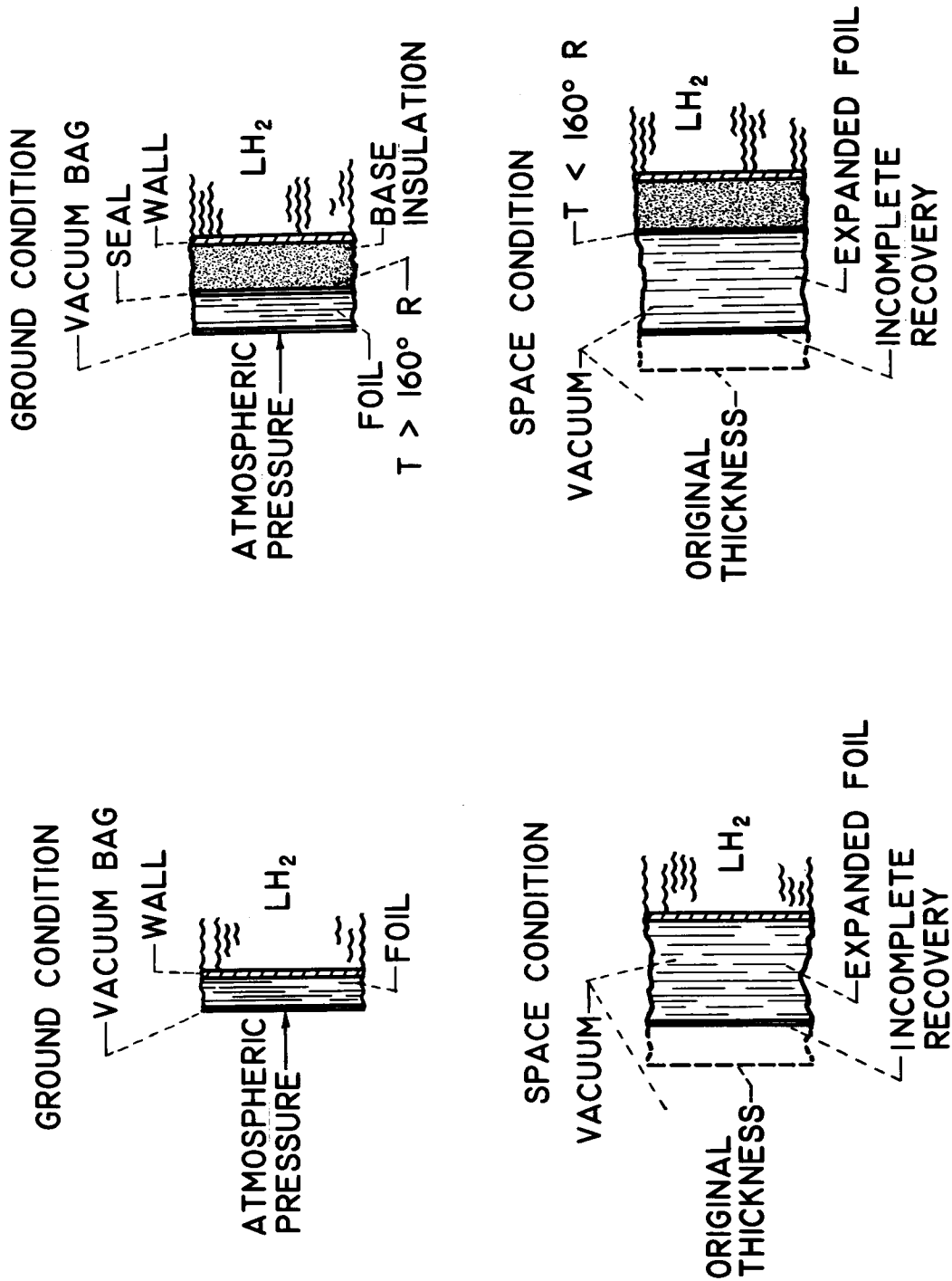
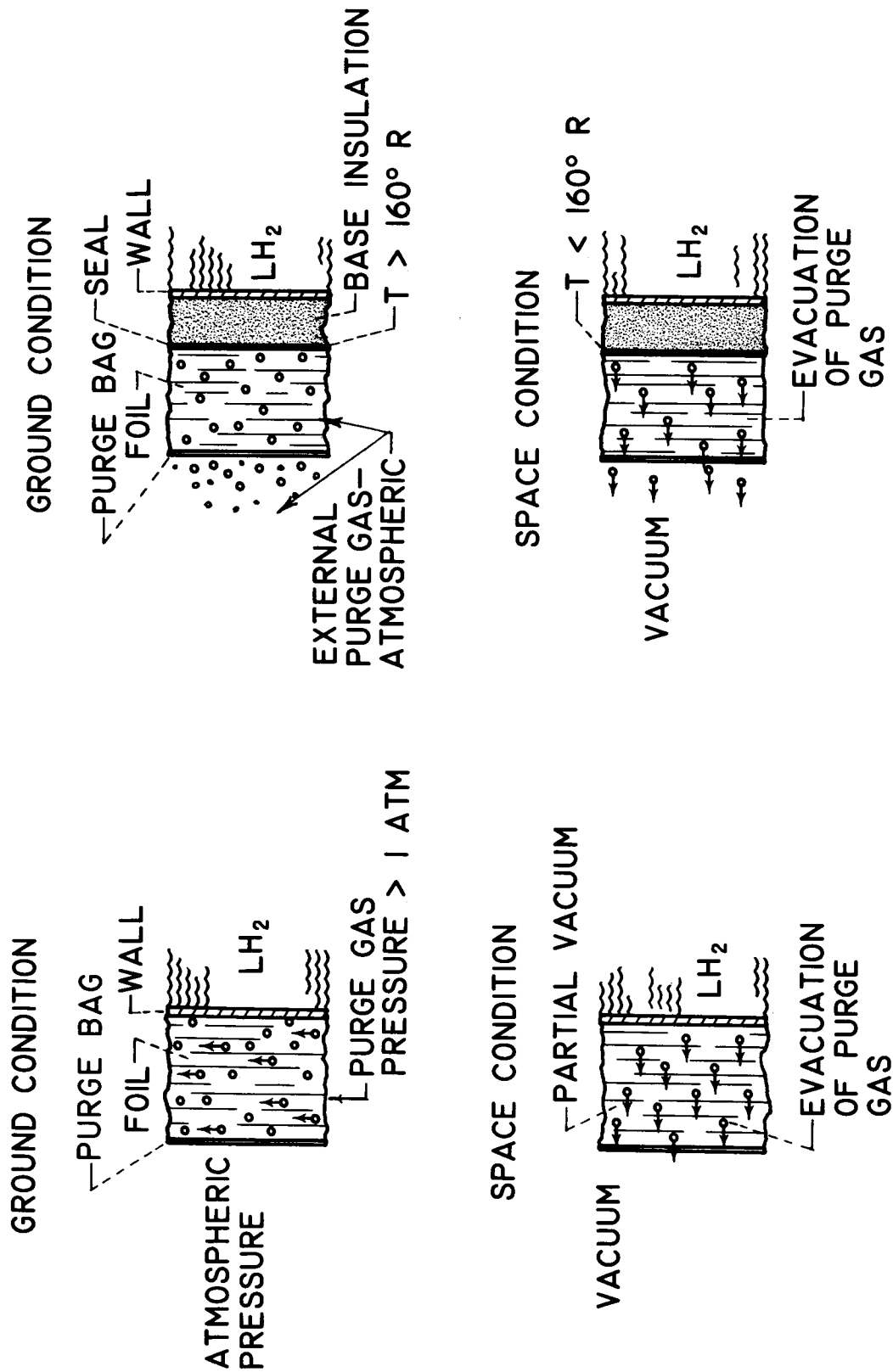


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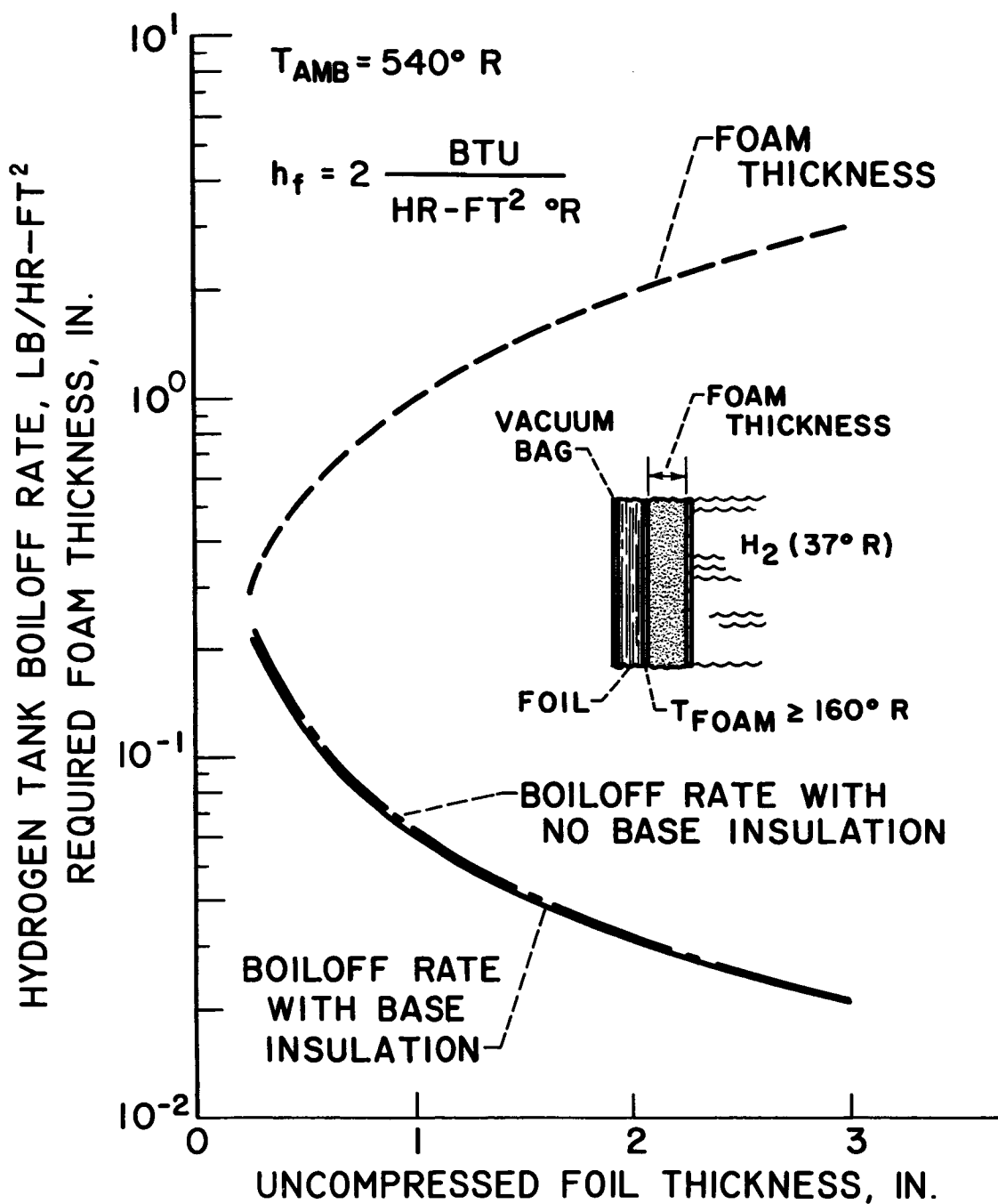


Fig. 9. - Effect of uncompressed foil thickness on hydrogen boilloff rate and required thickness of foam.

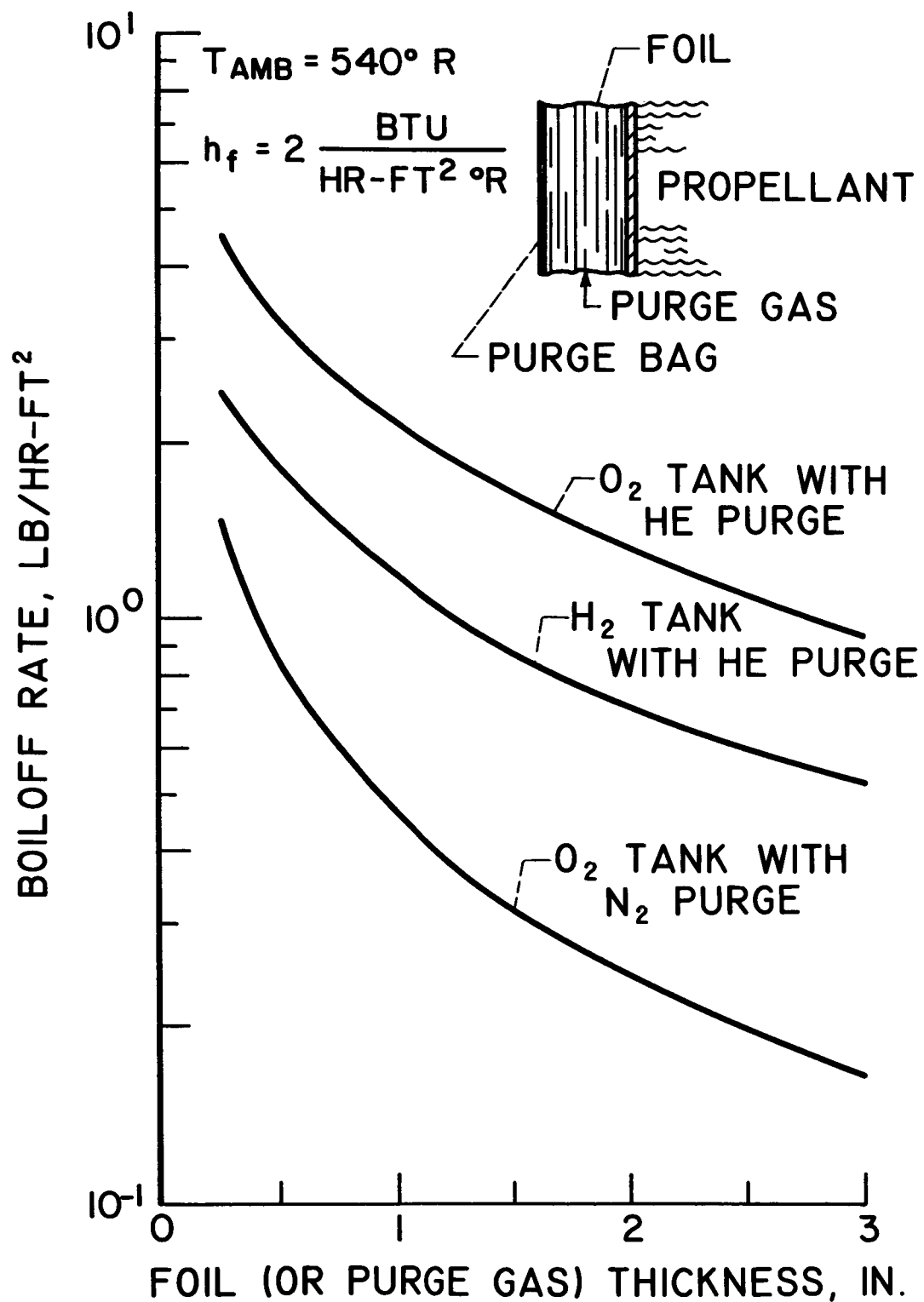


Fig. 10. - Effect of purge gas and foil insulation thickness on boiloff rates.

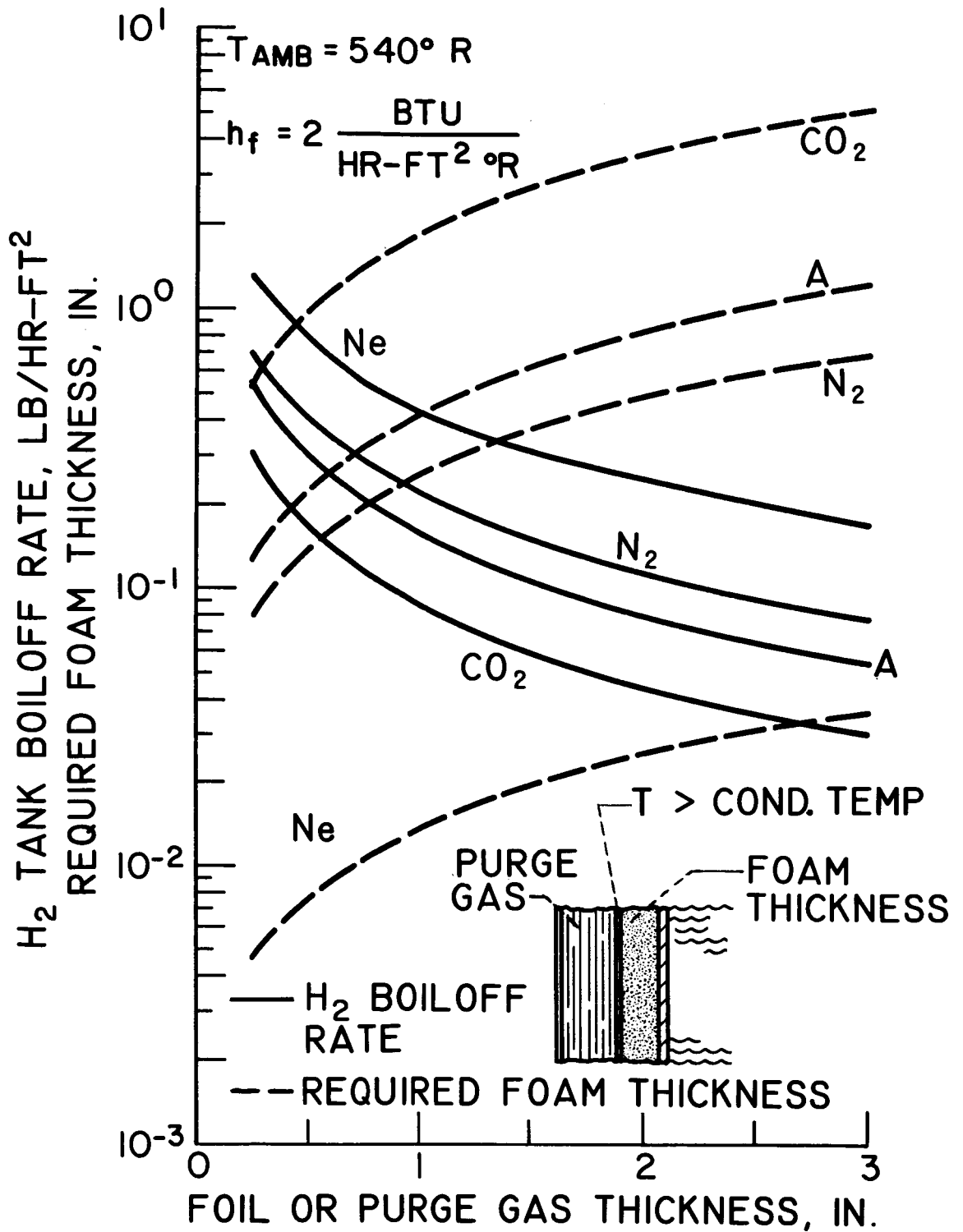


Fig. 11. - Effect of foil thickness on  $H_2$  boiloff rate and required foam thickness for various purge gases.

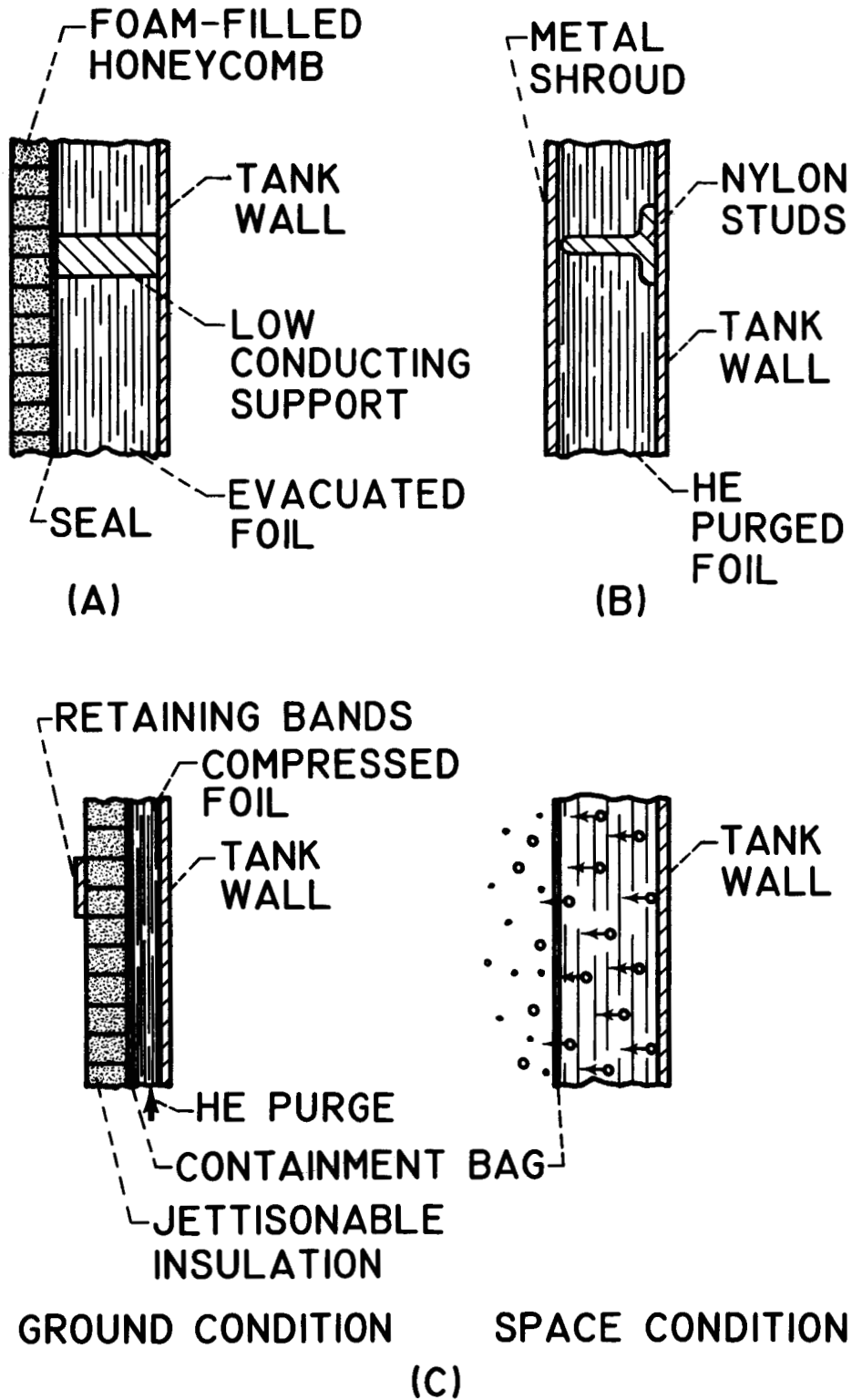


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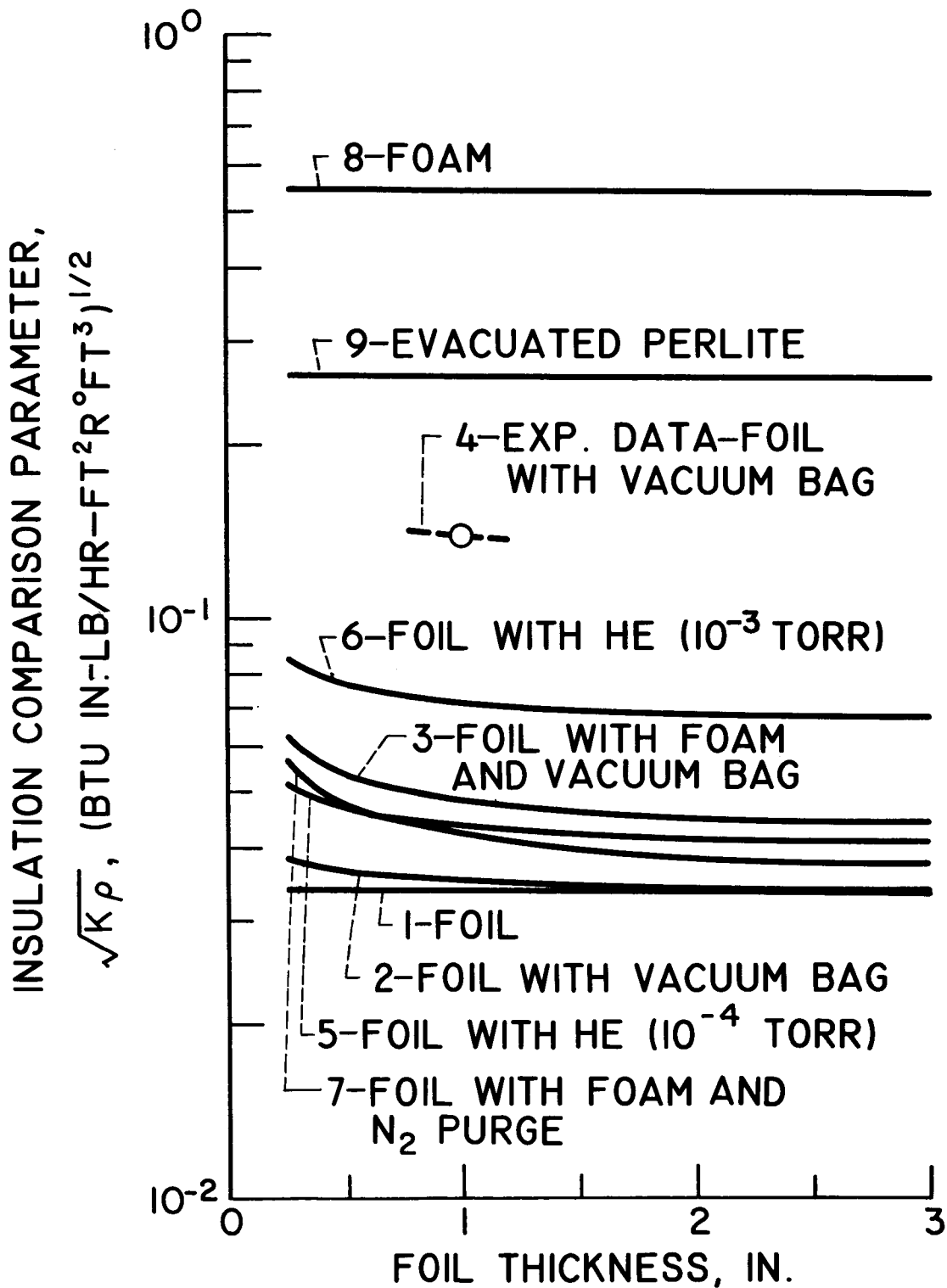


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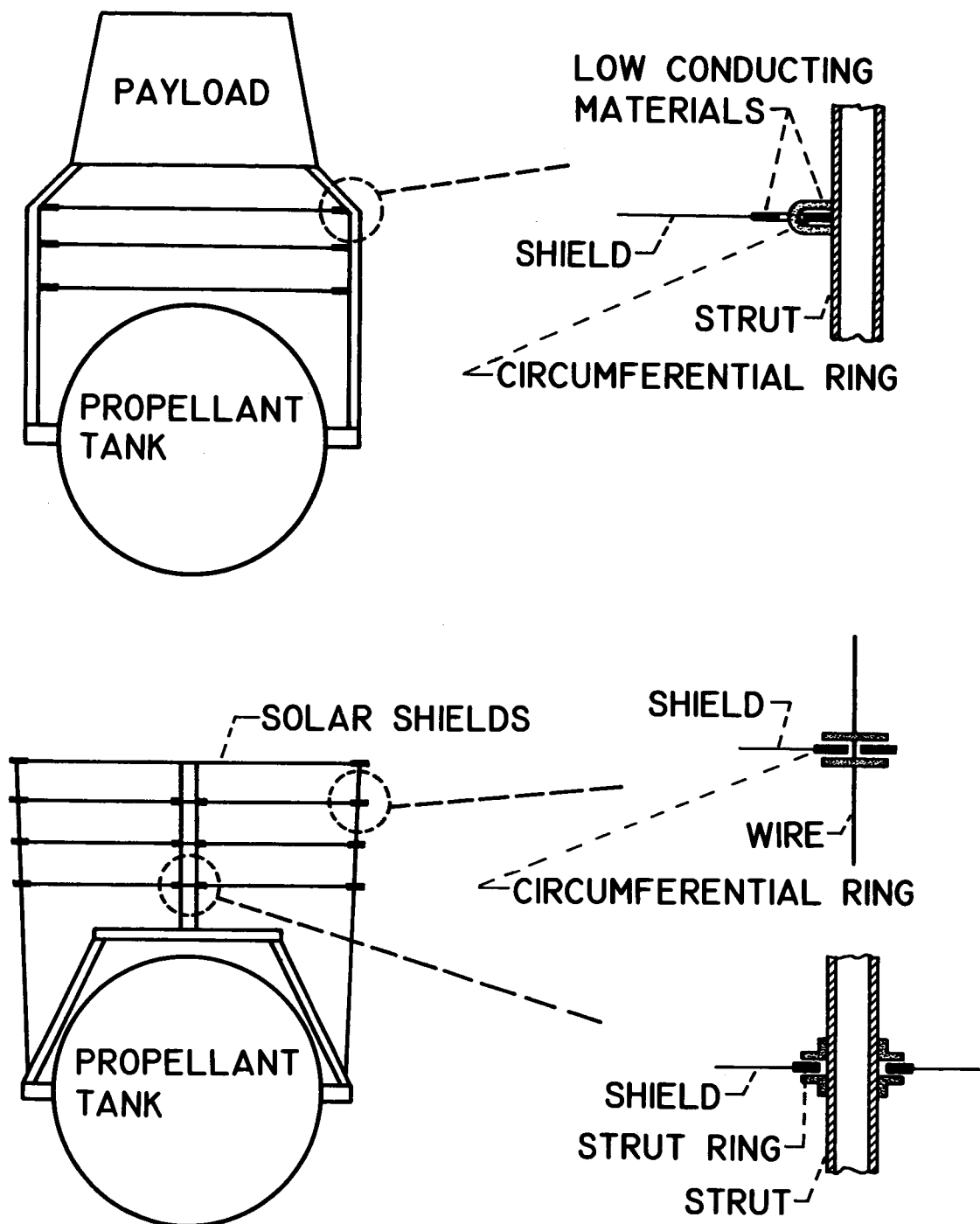
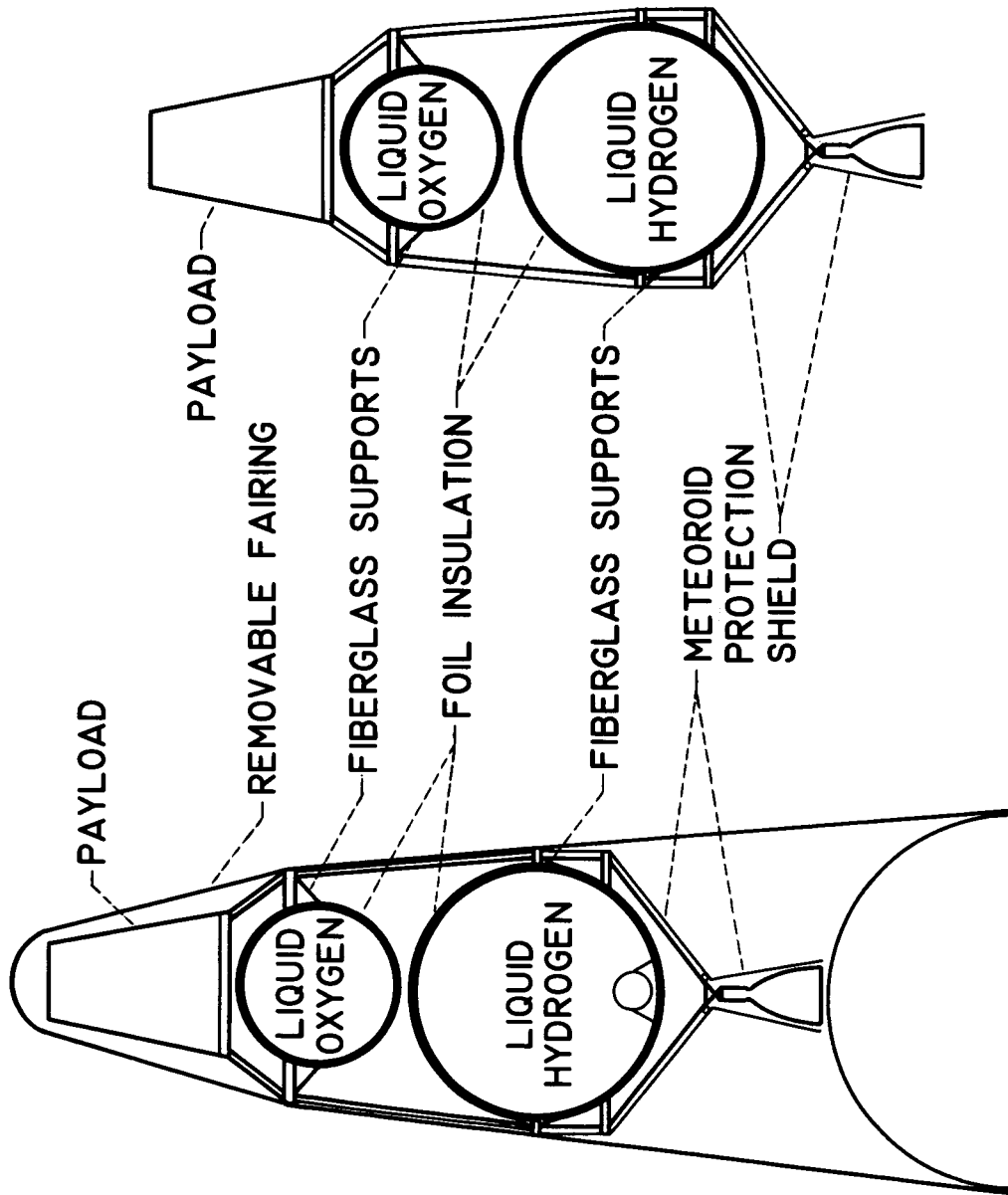


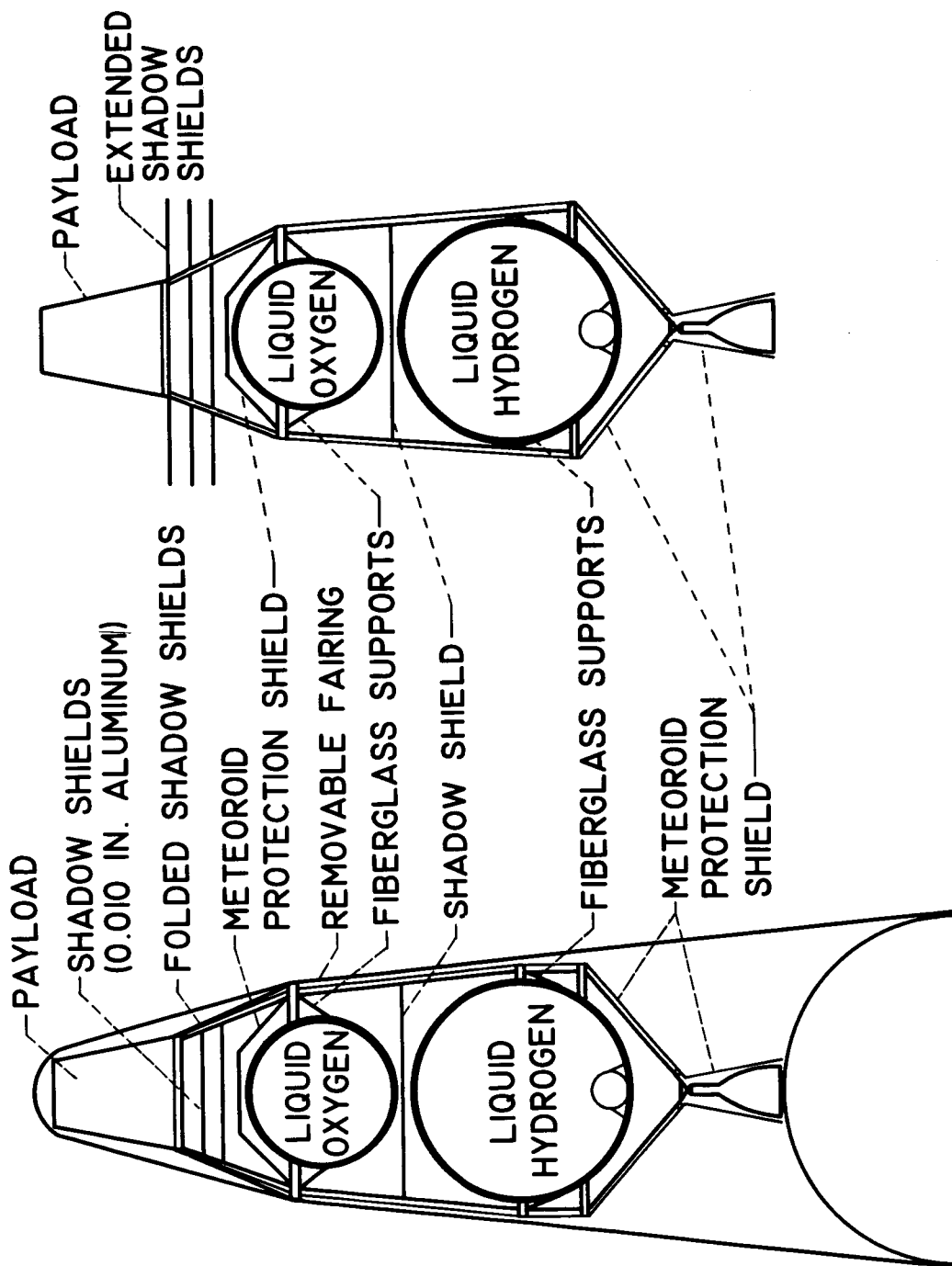
Fig. 14. - Low conducting shadow shield supports.



(a) Boost from Earth.

(b) Earth-Venus transit.

Fig. 15. - Venus orbiter with foil insulation.



(a) Boost from Earth.  
(b) Earth-Venus transit.  
Fig. 16. - Venus orbiter with shadow shields.